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Summary Engineering Report on the development of a ONE PLACE INFLATOPLANE,

contract NOnr 1860(00)

GER 8146

GOODYEAR AIRCRAFT

57 AA UNCLASSIFIED [redacted]

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R.C. Jones
~~[illegible]~~

Date: [illegible] By direction of Chief of Naval Research (Code 461)

Summary

Engineering Report on the Development of a One-Place

INFLATOPLANE

15 April 1957

GER 8146

Contract NOnr 1860(00)

GOODYEAR AIRCRAFT

57A [redacted] 3F

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SUMMARY

The design, development, fabrication, physical evaluation and flight testing of a one-place collapsible pneumatic aircraft are presented in this report. Performance of the Inflatoplane was excellent and substantially exceeded original estimates. It was found possible to collapse the Inflatoplane into approximately a 3 x 3 x 4 foot package and one man was able to move this package reasonable distances and assemble the airplane unaided in a short period of time. The future development of the Inflatoplane and the fabrication of additional field evaluation models are strongly recommended.

INTRODUCTION

The need exists for a better means of escape for fliers downed in enemy territory. Present methods are generally limited in range, involve considerable risk to a number of persons other than the downed flier, and require perfect coordination between the rescuer and the rescued. It is this contractor's belief that downed pilots need a self-contained means of escape with considerable range which can be hidden and used at the most opportune time.

Such a device is a collapsible aircraft utilizing pneumatic construction which could be parachuted to a downed flier, moved to a suitable take-off site, and rapidly prepared for flight. In order to meet these general requirements, the aircraft also had to meet specific design requirements which included the following items:

1. Empty weight not to exceed 170 lbs.
2. 240 lb payload including the pilot
3. 60 knot top speed
4. 4 hour endurance at cruise speed
5. Consideration to keeping engine noise to a minimum
6. Powered by a 40 hp Nelson engine or equivalent

In addition to these specific requirements consideration was to be given to a source of air resupply in the event of damage by small-arms fire, keeping the package airplane size to a minimum, dropping by parachute, and handling of the

airplane on the ground and preparation for flight (reference 1).

The possibility of making an efficient aircraft structure utilizing pneumatic construction became a reality several years ago with the development of "Airmat" by the Goodyear Tire and Rubber Company. "Airmat" is a pneumatic structural material consisting of two layers of fabric restrained, when inflated, by continuous tie yarns dropped from one layer of fabric to the other at the time of weaving. "Airmat" cloth is coated on the outer sides with neoprene and an extra layer of fabric to make it gas tight and give added strength in the desired directions. "Airmat" has been made in flat panels up to 15 inches thick and has been successfully used as structural members in radomes, 16-foot powered boats, arctic maintenance shelters, and many other items. Realizing the possibilities of this new material in the aircraft field Goodyear Aircraft Corporation fabricated an "Airmat" airfoil section. This section was tested statically in a wind tunnel to determine its suitability for use in helicopter rotor blades and airplane wings (references 2 & 3).

Based on the results of these tests Goodyear Aircraft Corporation proposed to the military services the fabrication of an all pneumatic aircraft. Considerable skepticism was expressed as to the possibility of successfully making such an aircraft. To overcome these misgivings Goodyear Aircraft Corporation decided to attempt the fabrication of a crude model of a small one-place aircraft (reference 4). In this limited program it was not possible to develop weaving machinery to

make a pneumatic airfoil and the wing airfoil was made of flat panels bent into an airfoil shape. Other short cuts also had to be taken which resulted in a number of components being heavier than necessary and aerodynamically "dirty." Although flight performance was marginal the point was definitely made that pneumatic construction could be used successfully in the major structural members of an airplane.

With this airplane successfully demonstrated the present program, sponsored by the Office of Naval Research (ONR), was initiated to improve the performance and decrease the weight of the one-place Inflatoplane and develop machinery for the weaving of an airmat airfoil-shaped wing.

The program for the development of the one-place Inflatoplane was divided into four phases as follows:

I PRELIMINARY DESIGN

II DETAILED DESIGN

III PHYSICAL EVALUATION

IV FLIGHT TEST

A study of the packaging and ground handling of this aircraft will be made as part of Phase I in the Two-Place Inflatoplane Development Program.

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Section 1

PHASE I

PRELIMINARY DESIGN

GER 8146

Leona K. Blair
Charles J. Carey
W. D. McDermott
J. T. Blair

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SELECTION OF THE BASIC CONFIGURATION

The basic configuration selected was the same as that used with the Goodyear Aircraft Corporation Inflatoplane and proposed in reference 1. This configuration consisted of a high wing monoplane with the engine mounted above the trailing edge of the wing, the cockpit located ahead of the fuselage, a single wheel landing gear, and conventional tail surfaces. This configuration was selected to give good aerodynamic performance and the simplest structure, thereby reducing the weight to a minimum and providing the smallest packaged size. The general advantages of this configuration can be summarized as follows:

1. Engine: With the engine mounted on a pedestal above the trailing edge of the wing, the drag on the airplane from the high speed slip stream from the propeller was reduced to a minimum. This was particularly important in this type of airplane as there were many exposed cables and fittings creating drag along the fuselage. A long landing gear was not required for propeller ground clearance. This was particularly important in reducing the packaged size of the airplane. Also, the wind and fuselage cone provided an excellent base for the pedestal mount.

The high position of the engine also served as a good attachment point for the brace wires supporting the wind under negative loadings.

2. Cockpit - Fuselage: By locating the cockpit ahead of the fuselage the fuselage

construction was greatly simplified and proper balance of the airplane was easily maintained. This was particularly important since a continuous fuselage cone provided an excellent base for mounting all the main structural members of the airplane and keeping them properly aligned. The cockpit was enclosed to reduce aerodynamic drag and to give pilot protection during long flights.

3. Wing: The high wing position was chosen to give good stability and a good angle for the main, lower bracing cables.

4. Landing Gear: The selection of single wheel landing gear ties in closely with the pedestal mounted engine and high wing configuration, as well as with packaging, weight and performance factors. The elevated position of the engine and wing reduced the necessary length of the landing gear to the minimum required for fuselage ground clearance. This favored the selection of a unicycle gear and greatly reduced the structural requirements. This in turn kept weight, aerodynamic drag, and packaged size of the landing gear to a minimum.

5. Empennage: From experience gained with the Goodyear Aircraft Corporation Inflatoplane it was found that conventional tail surfaces were the easiest to brace as well as fabricate.

REFINEMENT OF THE BASIC CONFIGURATION

In establishing the over-all dimensions of the various components primary importance was placed on keeping an equal, low inflation pressure for all inflated components while insuring adequate performance and keeping

the size and number of rigid components to a minimum. The low inflation pressure was necessary to reduce air loss to a minimum in case of damage and keep down the weight of gum required in the fabric for air retention. A uniform inflation pressure for all pneumatic components simplified the compressor and air control system. A compromise had to be made between performance and a low inflation pressure since the low pressure favored thick, short members while aerodynamic performance was improved by the use of thin, relatively long components. Packaging requirements dictated the elimination of as many rigid parts as possible.

For preliminary design purposes the gross weight of the airplane was established as follows:

Airplane, empty	170 lbs
Payload	240 lbs
Fuel (20 gal.)	130 lbs
Gross Weight	540 lbs

Top speed was estimated at 60 knots. With the establishment of these basic requirements the preliminary design of the various components commenced.

1. Wing: Preliminary design of the wing has as its purpose the selection of the proper airfoil, aspect ratio, wing area, determination of the structural properties of the section through static tests on flat airmat panels, and fabrication and test of a full scale wing.

a. Selection of Airfoil, Aspect Ratio, and Wing Area: Weaving limitations and

the nature of the inflated wing structure dictated the selection of a symmetrical airfoil. The width of the main section of the wing was limited to 52 inches or less because of the width of the standard loom. To keep the operating pressure low a fairly thick airfoil was required. However, the curvature of the upper and lower surfaces of the wing had to be kept down so that the weaving shuttle would not be required to move through an excessively curved path. With all these factors in mind an NACA 0015 airfoil was selected as a good compromise between weaving limitations, low profile drag, and high structural strength at low inflation pressures. Figure 1 shows the wing lift coefficient vs. angle of attack for this airfoil.

The requirement of low inflation pressure and structural rigidity dictated the choice of the lowest aspect ratio consistent with satisfactory performance. Initially an aspect ratio of 4.4 was selected as a good compromise even though a larger aspect ratio would have resulted in somewhat improved performance. Figure 2 shows the total wing drag at an airspeed of 60 knots as a function of wing area and aspect ratio for a gross weight of 550 pounds. The same figure also shows the wing area required as a function of take-off speed and aspect ratio at the same gross weight. A wing area of 110 square feet was chosen to give a take-off speed of approximately 40 knots at a maximum gross weight of 550 pounds. This area is also approximately the area which gives maximum total wing drag at 60 knots with the selected aspect ratio of 4.4.

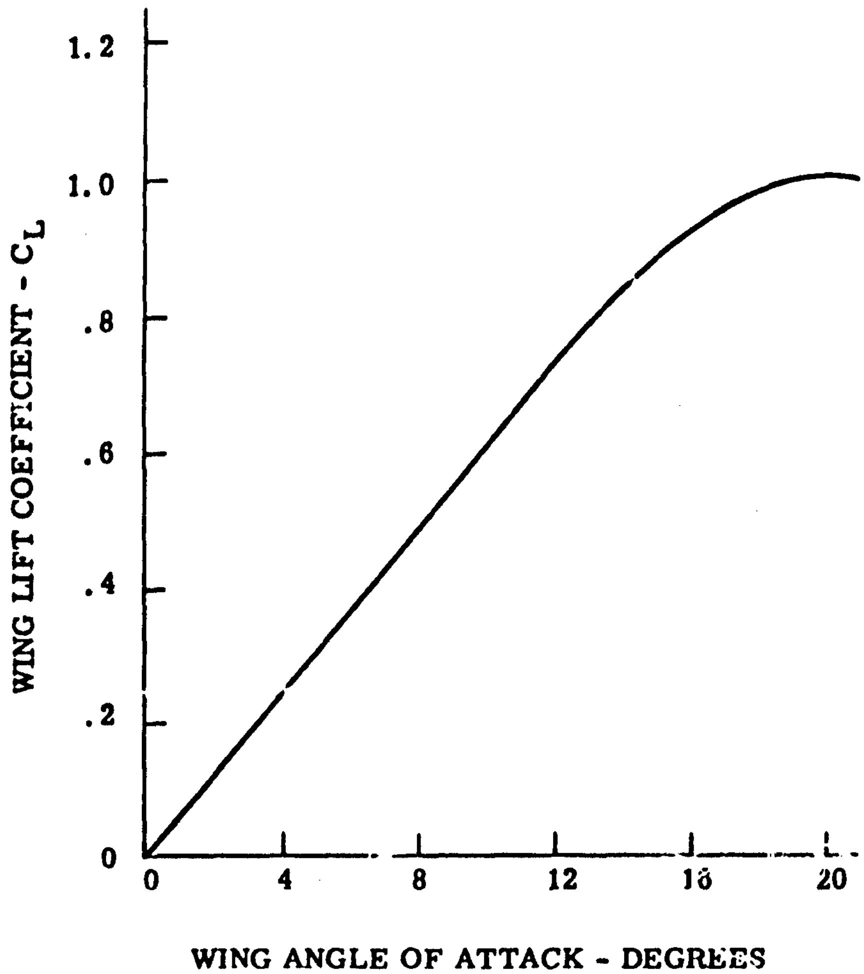
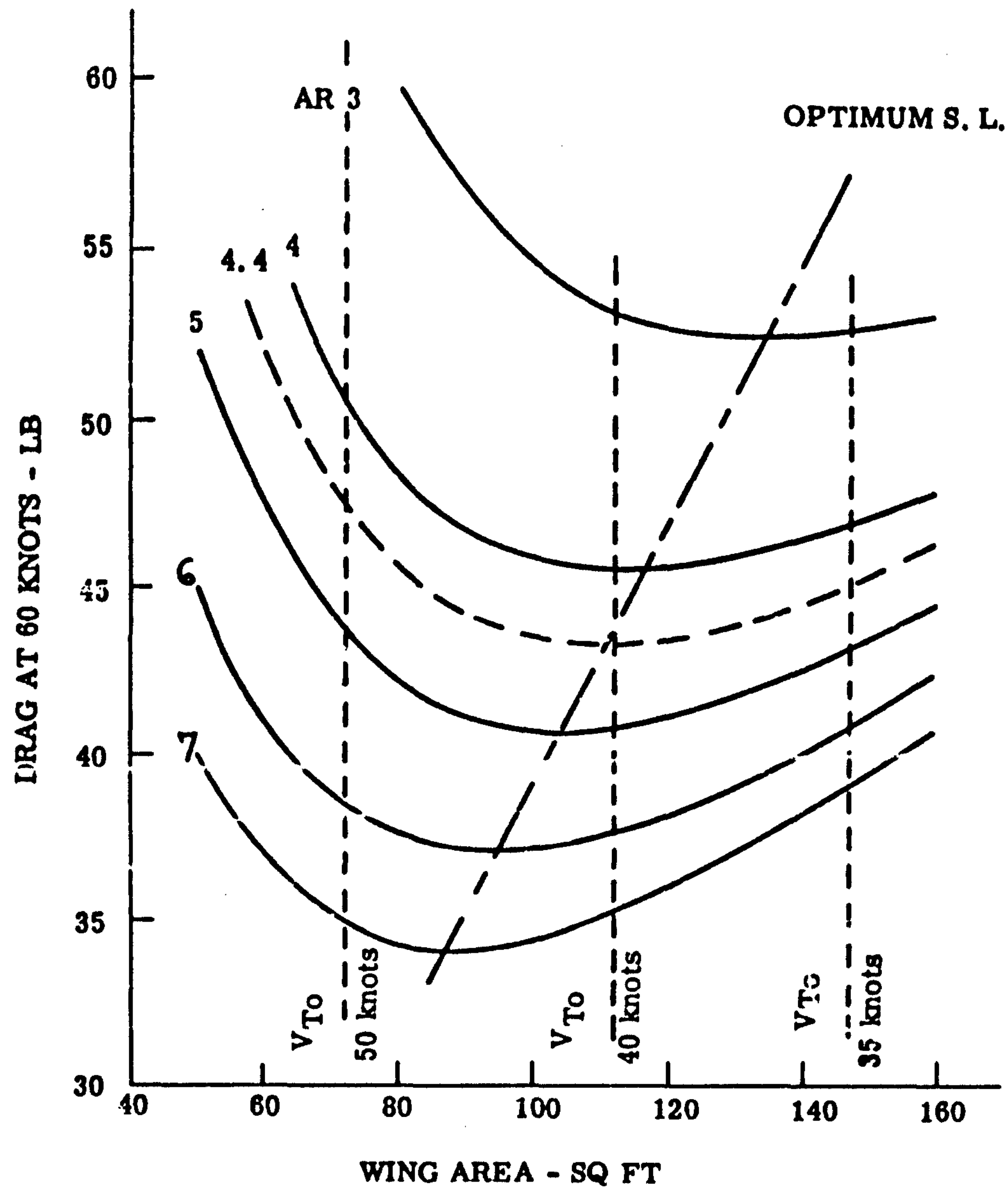


Figure 1. Wing Lift Coefficient vs Angle of Attack



GROSS WEIGHT - 550 LB
AIRFOIL - NACA 0015
DRAG COEFFICIENT - 0.016

Figure 2. Total Wing Drag at 60 Knots vs Wing Area and Aspect Ratio

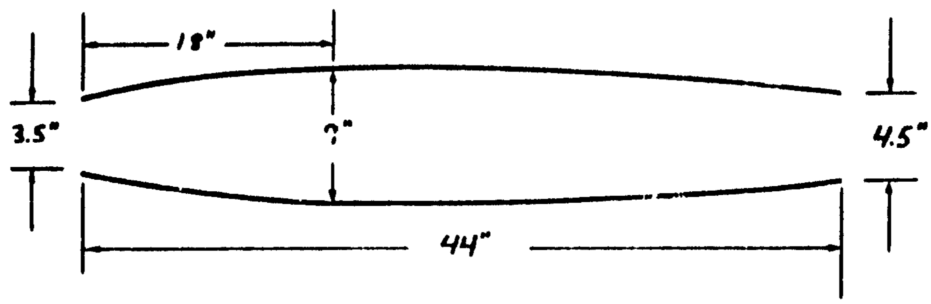
Structural analysis indicated that the necessary inflation pressure of the wing for an aspect ratio of 4.4 would be about 6.5 psi, while the pressure would be 10 psi for an aspect ratio of 6. An inflation pressure of 6.5 was found to be consistent with pressure requirements in the fuselage and tail surfaces, while a pressure of 10 psi would be unnecessarily high and increase the fabric strength requirements of these other members. Concurrently, tests were conducted to determine the air loss through bullet hole punctures in a 3-inch "Airmat" at various internal pressure (appendix A). The results of these tests indicated that the losses at 10 psi were excessive from the standpoint of compressor requirements and that the internal pressure should be kept to 7 psi or less. With all these considerations in mind the wing dimensions were established as follows:

Aspect Ratio	4.4
Airfoil	NACA 0015
Wing Area	110 sq. ft.

b. Flat Panel Tests: Simple bending tests were conducted on one 2-inch and one 3-inch thick "Airmat" panel 2 feet wide by 10 feet long at different inflation pressures to determine various structural properties which could be expected in the wing. These tests are consolidated in a later section of this report entitled Phase III - Physical Evaluation. Torsion tests on a flat 3-inch panel with and without bias plies were also conducted and are reported under the same section.

c. Fabrication and Test of a Full-Scale Wing: Fabrication of the full-scale

wing was subcontracted to the Goodyear Tire and Rubber Company. The actual weaving of the "Airmat" wing cloth in turn was subcontracted to Ellenboro Mills, Ellenboro, North Carolina, which has produced "Airmat" cloth for several years. Normally, "Airmat" cloth is woven in flat panels where the shuttle is free to follow a straight path through the shed, or separated warp yars. In weaving a contoured airfoil, however, the shed is curved in the airfoil shape which tends to throw the shuttle off its straight path and malfunction. The curvature that had to be woven for the main section of the wing is shown below.



To reduce this curvature the "false pick" method of weaving was employed. In this method the thickness in weaving is reduced by 1/3 by the use of extra fill yarns. After weaving, the extra fill yarns or "false picks", are removed and the cloth may be pulled out to its full thickness. With the curvature reduced by 1/3 the shed was opened wide enough so that the shuttle could pass through a relatively straight path. The same procedure was used in weaving

the aileron and flap section. Some difficulty was encountered in using a light 70/2 denier fill yarn and a heavier 210 denier yarn had to be used, resulting in greater strength in the fill direction than necessary.

The wing cloth was coated with neoprene using standard production equipment and a single bias ply was applied by machine. When inflated the first wing took on a uniform twist throughout the span. This twist was attributed to differential elongation in the warp and fill directions of the cover ply. A second wing was then made with two bias plies at 90 degrees to each other. This resulted in a straight wing (See figure 3). The bias ply on the wing was necessary to resist torsional stresses. The aileron-empennage material had only a single straight cover ply since torsional stresses in these members were small.

Static bending tests were conducted on the first test wing. The results of these tests are incorporated in the section entitled Phase III - Physical Evaluation.

2. Fuselage - Empennage: As an initial trial, the tail moment arm was set at three times the length of the wing chord, giving an over-all fuselage length of 17 feet. The diameter of the fuselage at the cockpit was set at 24-inches based on the approximate width of the cockpit. A 14-inch diameter for the tail end of the cone was selected based on experience with the Goodyear Aircraft Corporation Inflatoplane. The horizontal tail surface area was set at 18.2 percent of the wing area and the vertical surface at 13.8 percent of the wing area. The maximum downward tail load was estimated to be 230 pounds. Preliminary stress analysis showed that the fuselage would not be adequate to support this

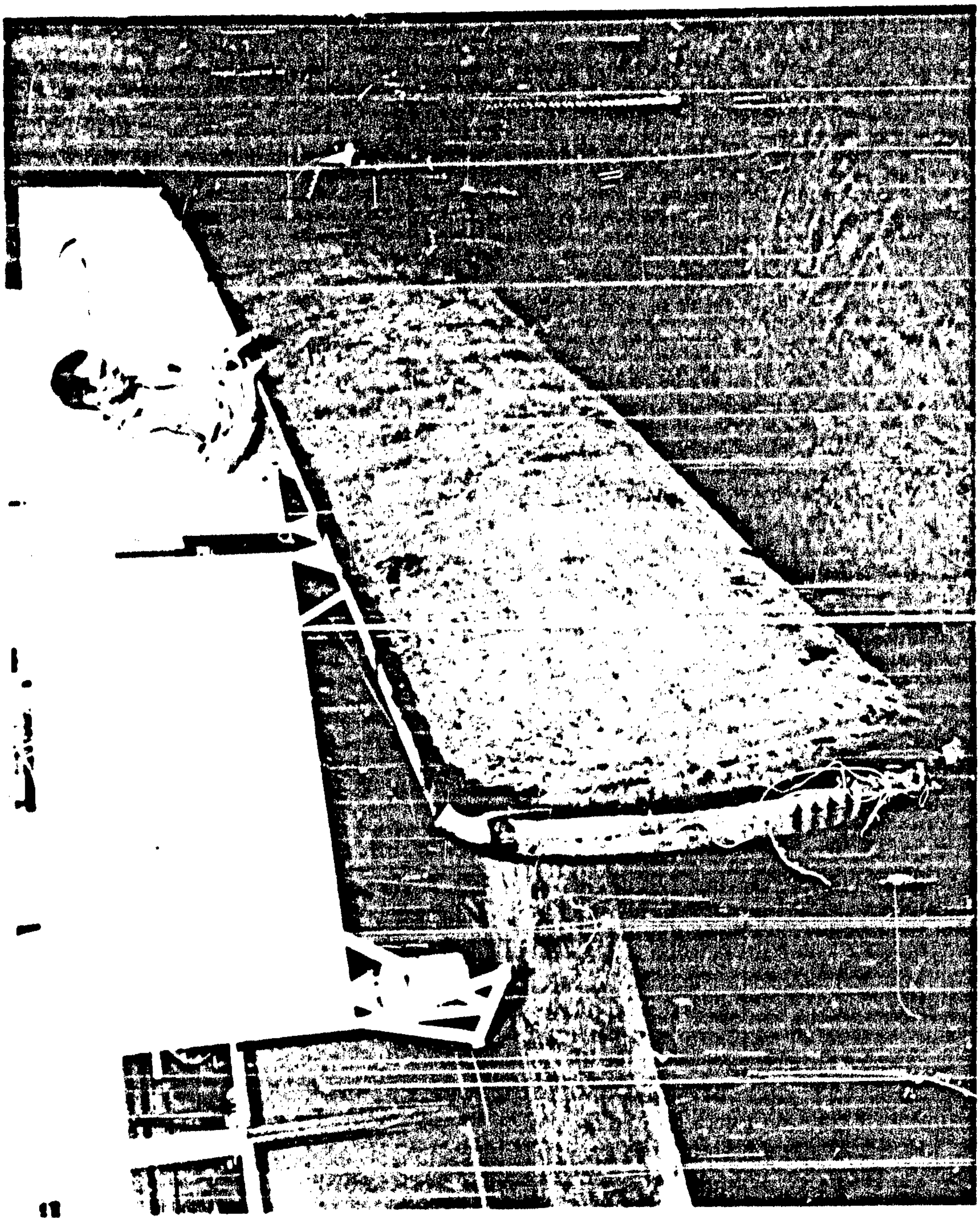


Figure 3. Main Wing Section

load at a 7 psi inflation pressure. Analysis of the longitudinal stability of airplane and weight and balance studies for different pilot weights indicated that the tail moment arm or the area of the tail surfaces could be reduced and still maintain longitudinal stability. It was found more advantageous from the standpoint of fuselage deflections to shorten the fuselage rather than decrease the area of the tail surfaces.

Further analysis of the stresses, stability, and balance resulted in increasing the maximum diameter of the fuselage to 27 inches, shortening the fuselage 15 inches, and adding a brace wire from the rear of the engine mount to the tail. Because the high thrust line aggravated the down load on the elevator, a tail brace wire was required to resist the down load. No brace wire was required to resist the up load, which was of considerably smaller magnitude. With the fuselage dimensions and inflation pressure established it was possible to select the fuselage material. A 2-ply nylon-neoprene fabric was designed to give the lightest possible fabric adequate to carry the loads. Cotton, fortisan, and dacron fabrics were also considered but discarded for various reasons. Cotton fabrics were easily obtainable and had better elongation properties than nylon but had less strength per unit weight. Fortisan fabrics had a high modulus of elasticity and great strength per unit weight, but showed a tendency to lose strength when folded or creased. Dacron had all the advantages of nylon, high strength/weight ratio, abrasion resistance, good time-load characteristics, and was considerably stiffer than nylon but was not available in sufficient quantity in the time available.

Preliminary analysis of the tail surfaces showed that the thickness should be 3-3/8 inches at the 7 psi inflation pressure. With this thickness it was possible to use the same material as was used in the ailerons and taper the trailing edges of the elevator and rudder. A V-tail and a tail with the horizontal stabilizer mounted up on the fin were considered in addition to the conventional configuration and were rejected because they complicated the bracing arrangements without improving performance or control.

3. Cockpit: The cockpit was designed to carry a 240 pound, 6 foot 4 inch pilot. Emphasis in preliminary design was placed on keeping the frontal area of the cockpit to a minimum to reduce drag. A number of different seating attitudes for the pilot were investigated with this purpose in mind. Due to the nature of "Airmat" construction it was also necessary to keep the design as simple to build as possible.

4. Landing Gear: A single wheel landing gear with wing tip skids was selected to give a minimum number of rigid parts in packaging and low aerodynamic drag. This gear was originally conceived as a fiberglass well form fitted and anchored to the forward hemispherical end of the fuselage, but was later changed to an aluminum tubular structure. It was decided that a readily detachable tricycle landing gear would also be provided for training purposes.

5. Fuel System: To eliminate the need for a fuel pump and the accompanying electrical power source it was decided to use a fuel cell pressurized by the airplane inflation air. In order to maintain proper airplane balance, the fuel

cell had to be located on or near the center of gravity. With the 4 hour endurance requirement the capacity of the fuel cell was set at 20 gallons based on limited knowledge of engine performance. It was found that the best location for a fuel cell of this size was inside the fuselage itself where it could be easily located exactly on the center of gravity. In this location a bladder type fuel cell could be used with the pressure inside the fuselage acting against the outside of the cell and forcing the fuel to the carburetor. This minimized the strength requirements of the fuel cell and made an extremely light and efficient system.

6. Engine Mount and Power Plant Assembly: A 40 hp Nelson engine was selected as the power plant since it was the lightest engine available which could deliver the necessary power. Experience with this engine on the Goodyear Aircraft Inflatoplane showed that it was also dependable. A 47-inch diameter, 12.5 degree pitch propeller was selected to give near optimum performance throughout the speed range. To reduce drag around the engine mount to a minimum, a single pylon type pedestal mount was chosen. A stress study was conducted to determine the best and lightest material which could be used. Steel, magnesium, titanium and aluminum were all studied as possible materials. Magnesium was found unsatisfactory since it had very low notch sensitivity resistance. It was also subject to excessive distortion during welding which would be difficult to control. Titanium was found to be too expensive and difficult to procure. Although a steel mount was found to be adequate structurally, its weight was almost double that of an aluminum mount. Aluminum construction was found to be structurally adequate and lightweight and was therefore selected.

Cowling of the engine was considered in order to reduce the considerable aerodynamic drag of the exposed engine. Several other aircraft companies using the engine were contacted as well as Mr. Ted Nelson, developer of the engine, to discuss their experiences with cowling the engine. Cowling of this engine has been attempted only in helicopter installations where no ram air was available for cooling. Mr. Nelson foresaw no difficulty in cowling a tractor installation of the engine. Further aerodynamic studies indicated that the 60 knot top speed could be met without cowling the engine and therefore the matter was dropped.

It was decided that a compressor should be selected which could resupply air lost through three .30 caliber bullet holes and maintain sufficient pressure for safe flight under a minimum flight condition. From bullet hole tests in a flat panel (Appendix A) it was found that this loss was 47 cfm at 3.5 psi internal pressure. Several pumps made by various companies were investigated and a vane-type compressor made by Pesco was selected. The Pesco Model 485 pump was found to supply the necessary quantity of air at the desired pressure and rpm.

The power requirements of this pump were low, the size and weight reasonable, and since the pump was in production it was readily available and reasonable in price. One disadvantage to this pump was that external lubrication was required. An oil supply and separator had to be included as accessories. A suitable pump which did not require lubrication had been designed by Pesco but had not been produced in quantities and was too expensive to procure for this program.

Several possible methods of driving the compressor were considered, including a direct drive off the rear of the crankshaft and belt-drive. Mr. Nelson was contacted again to give his opinion on this matter. He favored a belt drive off the propeller hub since the engine was not designed for power take-off at the rear of the crankshaft. Two main problems were presented in this method. One was torsional vibration effects and the other was the strength requirements of the pump drive. Both problems were complicated by the fact that the ignition breaker cam also had to be driven off the rear of the crankshaft. The disadvantage to the belt drive arrangement lay in the bulk of the system and the added drag around the engine. It was found possible to design a coupling between the compressor and rear of the crankshaft which did not interfere with the timing. This arrangement helped streamline the engine, thereby keeping the added drag to a minimum.

7. Flaps: The possibility of using flaps separately or in combination with depressed ailerons was studied to see what improvement in take-off performance could be realized. The following configurations were taken into consideration: Flap and drooped aileron deflected 15 degrees; and flap and aileron deflected 30 and 15 degrees respectively. (See figure 4).

Larger flap deflections increased the take-off distance; consequently, 30 degrees was approximately optimum.

For this investigation the aileron and flap chord were extended up to 6 inches aft of the present trailing edge; with corresponding aileron and flap chords

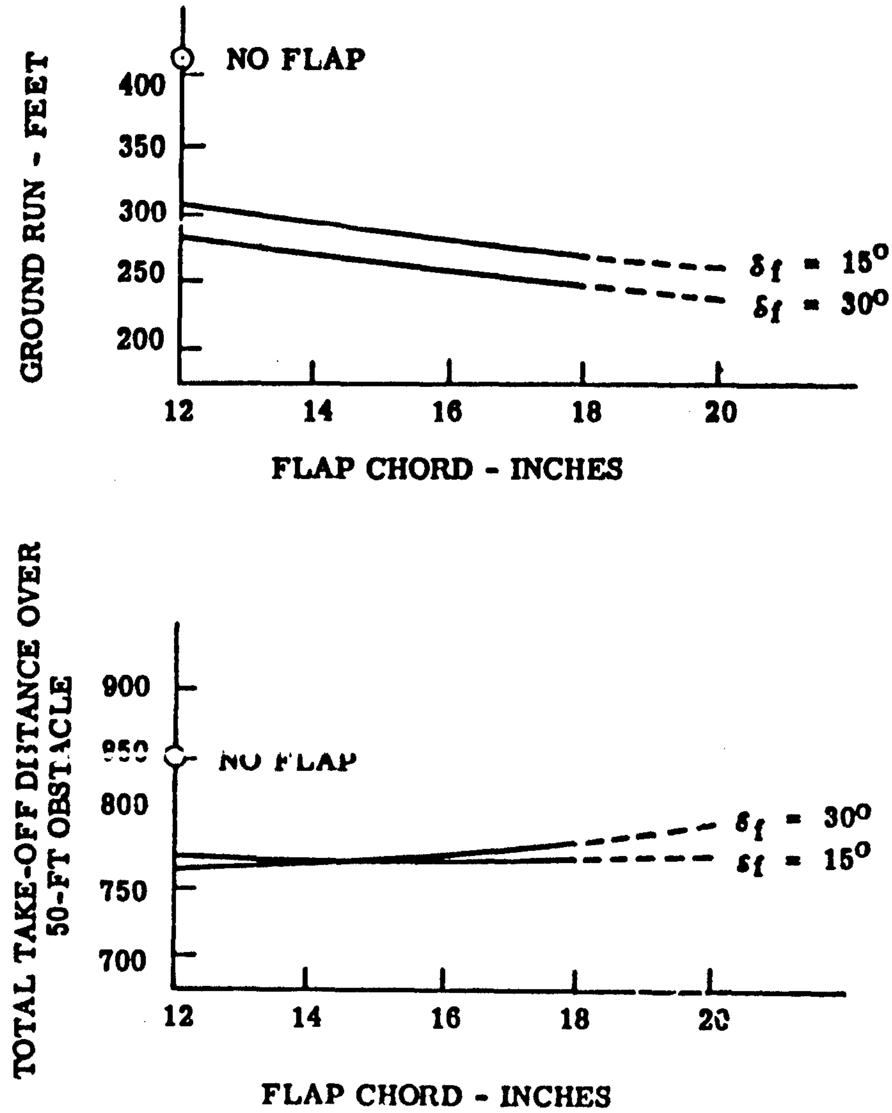


Figure 4. Take-Off Performance with Flaps

varying from 12 to 18 inches. These extensions varied the wing area from 110 to 121 square feet. The results from this investigation are presented in figures 4 and 5.

The ground run compared to a non-flapped wing was decreased by configuration (1) by approximately 110 to 140 feet and by configuration (2) by about 130 to 160 feet. The total take-off distance over a 50-foot obstacle was reduced to a lesser extent (up to 75 feet). The effect of extending the chord beyond that of a 12 inch flap chord showed that only a slight decrease in ground run could be attained, whereas total distance over a 50-foot obstacle was actually increased.

To utilize these flapped configurations, it was necessary to determine the tail area required for trim at a given c.g. location. It should be noted that all tails were considered geometrically similar, with the .25 chord at a constant body station, and not as designed for the present airplane. Extension of flap and aileron chord appreciably increased the tail area requirement (figure 5). Figure 5a presents c.g. location as a percentage of the extended mac. This means that the airplane c.g. must move aft with the extension of chord to maintain a constant percentage of mac. Figure 5b is presented to show the increase in tail area required to maintain a fixed airplane c.g. location. For large extensions and flap deflections, this requirement becomes excessive.

It can be seen from the figures that deflecting the 12-inch chord flaps and ailerons decreases the take-off ground distance 110 to 140 feet with a necessary

GROSS WEIGHT - 550 LB.
TAIL MOMENT ARM, ℓ_T - 147 INCHES (ORIGINAL)

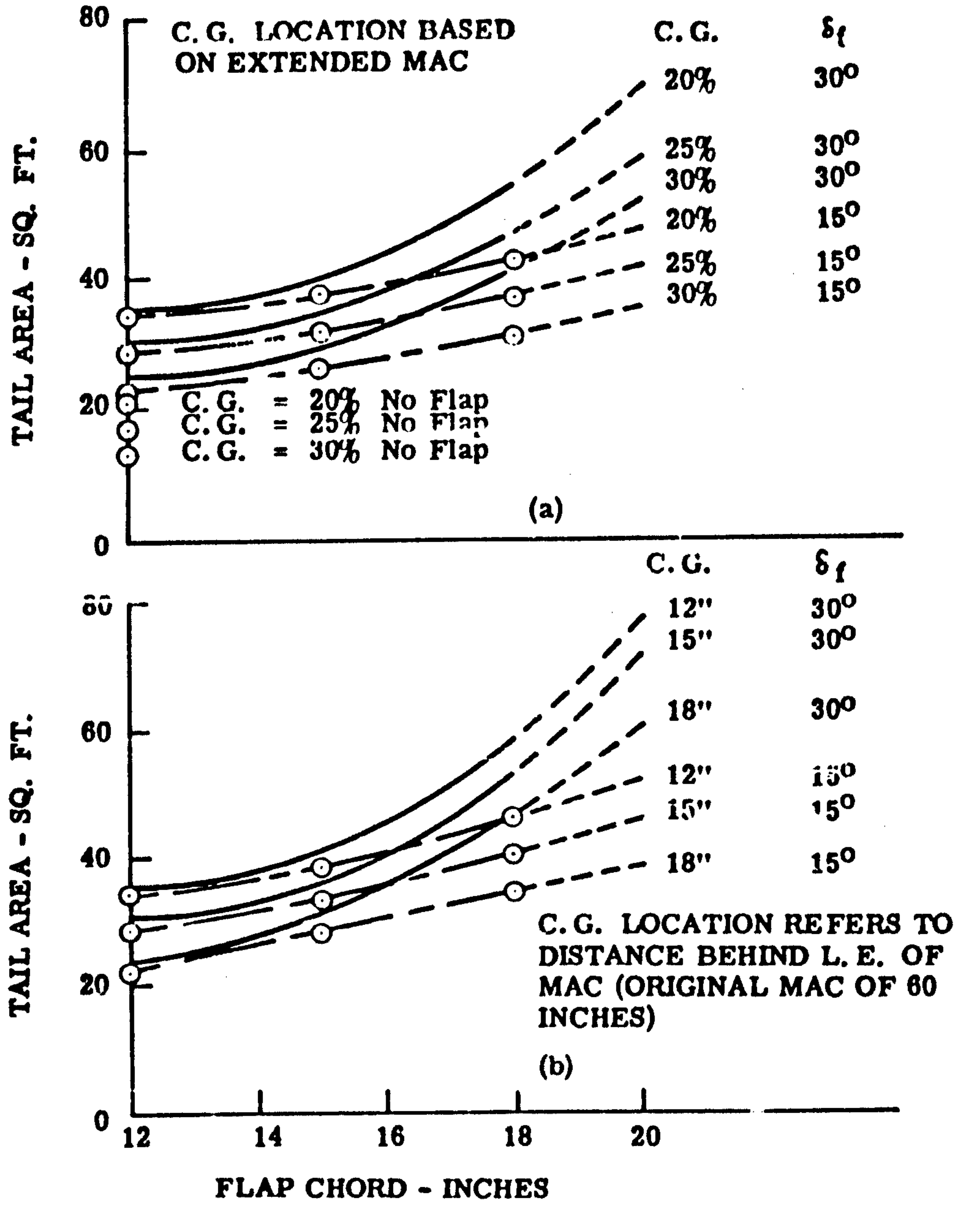


Figure 5. Tail Area Required to Trim For Take-Off With Flaps

increase in tail area of only 5 to 15 square feet depending on c.g. location. On the other hand, extending the chord of the flaps and ailerons required further increases in tail area in the same order of magnitude as required by flap deflection but only decreases the ground run an additional 20 to 30 feet.

Therefore, it appears that chord extensions are of little benefit on take-off performance considering the tail area penalties involved.

PRELIMINARY PERFORMANCE, STABILITY AND CONTROL ANALYSIS

After the establishment of the preliminary configuration an analysis was made to determine the performance, stability, control and airloads of this configuration. This analysis included an investigation of the effects of certain changes in the configuration which are noted in the appropriate sections. (See figure 6).

1. Configuration Characteristics:

a. Geometric Characteristics:

Wing	
Basic	
Total Area	110 sq. ft.
Span	22 ft.
Aspect Ratio	4.4
Taper Ratio	1.0
MAC	5.0 ft
Airfoil Section	NACA 0015
Aileron	
Type	Plain Flap
Span	9.50 ft. (Reduced to 7 ft.)
Chord (% wing chord)	20
Deflection Range	up 25° - down 15°

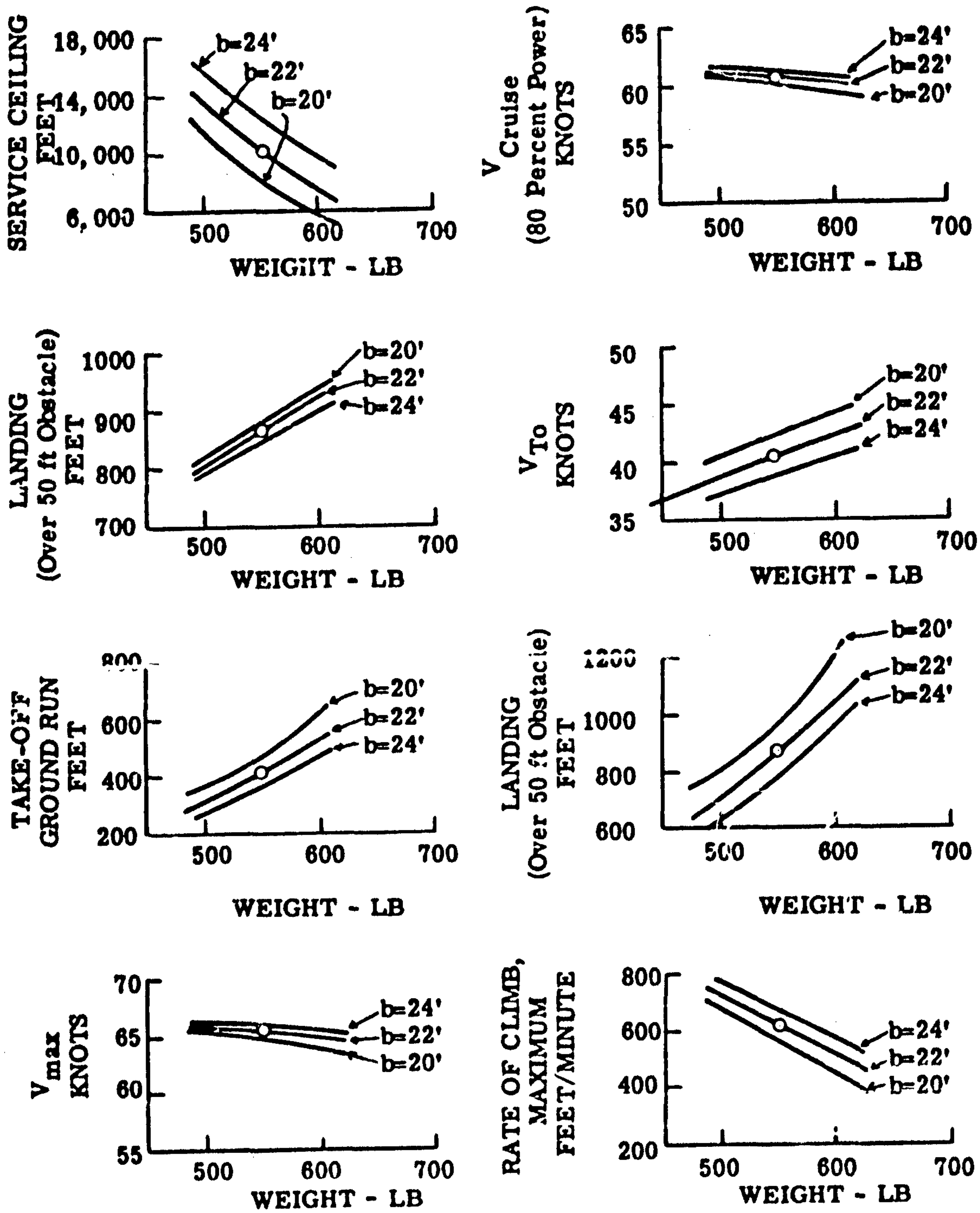


Figure 6. Preliminary Performance Estimate As a Function of Wing Span and Gross Weight (NACA Standard Atmosphere)

Horizontal Tail	
Basic	
Total Area	20 sq. ft.
Span	6.67 ft.
Aspect Ratio	2.22
Taper Ratio	.714
MAC	3.03 ft.
Elevator	
Type	Plain
Area	8.33 sq. ft.
Deflection Range	$\pm 30^{\circ}$
Vertical	
Basic	
Total Area	15.23 sq. ft.
Aspect Ratio	0.93
Rudder	
Type	Plain
Area	5.23 sq. ft.
Deflection	$\pm 30^{\circ}$
Fuselage	
Length	21.5 ft.*
Frontal Area	8.25 sq. ft.
Tail Moment Arms	
(Wing A.C. to Horiz. Tail A.C.)	13.9 ft.**

* Reduced to 19.83 ft.
** Reduced to 10.57 ft.

b. Weights:

Max. Gross Wt.	550 lbs.

c. Power Plant:

No. of engines	1
Model	Nelson H-59
Take-off BHP at RPM	40 at 4000*
Rated BHP at RPM	40 at 4000*

* Later information indicated a 44 HP rating.

2. Performance:

a. Drag Breakdown:

	Drag Area
Fuselage (8.25 sq. ft. at C_D = 0.22)	1.82
Wing (110 sq. ft. at C_D = 0.016)	1.82
Tail (35.23 sq. ft.@C_D = 0.021)	.75
Wheel (0.5 sq. ft.@C_D = 0.60)	.06
Engine (1.5 sq. ft.at $C_D \frac{qs}{qo}$ = 0.77)	1.16
Engine Pylon (1.0 sq. ft at $C_D \frac{qs}{qo}$ = 0.04)	.04
Wire Bracing and Control Cables	.75
Interference and Misc. Drag Items (20 percent of Total)	1.60
Total Drag Area	8.00 sq. ft.
Total Airplane C_{D_0} (based on wing area)	0.073

b. Lift: The estimated wing-lift curve is given in figure 1.

c. General Performance: Performance variations of service ceiling, landing distance, take-off ground run, maximum speed, cruise speed, take-off speed, take-off distance and rate-of-climb were estimated for three wing spans and three gross weights and are presented as a function of these parameters in figure 6 for a constant wing chord of 5 feet.

Due to the lack of information concerning the altitude performance of the Nelson H-59 engine, no estimates were made of the altitude performance other than service ceiling which was estimated from the Oswald performance charts given in reference 5.

3. Static Stability and Control:

a. Longitudinal Stability: A study of the static longitudinal stability and control yielded an aft c.g. limit at 39 percent mac and forward c.g. limit at 18 percent mac assuming a rigid structure. After establishment of the configuration, it was found desirable to reduce the bending moments on the fuselage due to tail loads by shortening the tail moment arm and/or reducing the horizontal tail area. Figure 7 shows the effect of shortening the tail moment arm on the c.g. limits. The effect of this change on the other stability parameters is indicated in the appropriate sections by arbitrarily shortening the tail arm 40 inches.

If the horizontal tail area had been reduced to 15 square feet with the preliminary tail moment arm of 167 inches, the aft c.g. limit would have been at 35 percent mac and the forward c.g. limit at 25 percent mac, assuming a rigid structure. Due to the anticipated flexibility of this inflated structure it appeared desirable to hold the c.g. as near the center of the stability region as possible. This is depicted by the dashed line evident in figure 4.

An estimation of c.g. shift with varying pilot weight was made by the weights analysis section and resulted in an 8 percent movement of c.g. This resulting movement of c.g. for the preliminary configuration is shown in figure 7 and further illustrated the need for a centralized position of the center of gravity.

b. Directional Stability: A study of the static directional stability yielded a more than adequate value of C_n of -0.00099 for the configuration. A desirable value

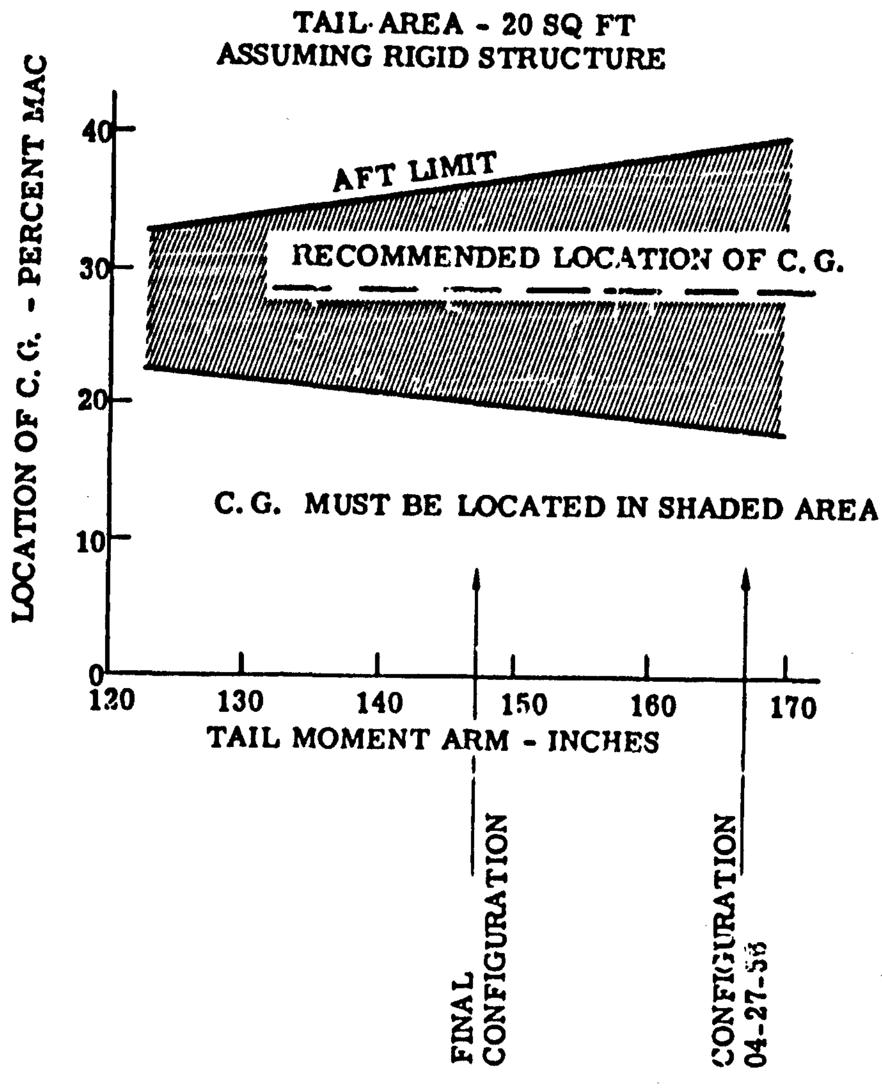


Figure 7. Allowable C. G. Range vs Tail Moment Arm (Distance From Wing A. C. To Horizontal Tail A. C.)

from reference 6 is:

$$C_n \text{ (Desirable)} = -.0005 \left(\frac{W}{b^2}\right)^{1/2} = -0.0053$$

If the tail moment arm had been shortened 40 inches C_n would be -0.00077.

An additional criterion requires that:

$$\frac{S_F\, l_T}{D^2\, L_F} = 0.4 \qquad \text{(ref 9)}$$

For the preliminary configuration:

$$\frac{S_F\, l_T}{D^2\, L_F} = 0.62$$

With the tail moment arm shortened by 40 inches:

$$\frac{S_F\, l_T}{D^2\, L_F} = 0.55$$

Where:

S_F = Fin Area

l_T = Tail moment arm

D = Maximum fuselage diameter

l_F = Fuselage length

c. Dihedral Effect: A study of the dihedral effect of the configuration yielded a value of C_l = 0.00064 with no geometric dihedral in the wing. This value is slightly higher than necessary but should not be detrimental to the response of the airplane. From reference 6 a desirable value is:

$$C_l = -0.5\ C_n = +0.000495 \qquad C_n = \text{initial configuration}$$

With the tail shortened 40 inches the desirable value would have been:

$$C_l = -0.5 \quad C_n = +0.000385$$

d. Lateral Control: A study of the aileron control power yielded a helix angle value of $\frac{pb}{2V} = 0.17$ for a rigid structure. Reference 7 gives a desirable value of $\frac{pb}{2V} = .12$ for a stalling speed of 40 to 45 knots. This desirable value could be obtained with an aileron span of 6 ft. if the structure were rigid.

4. Airloads: Brief studies were made to determine the loads acting on the airplane as a whole and also on some of the major components.

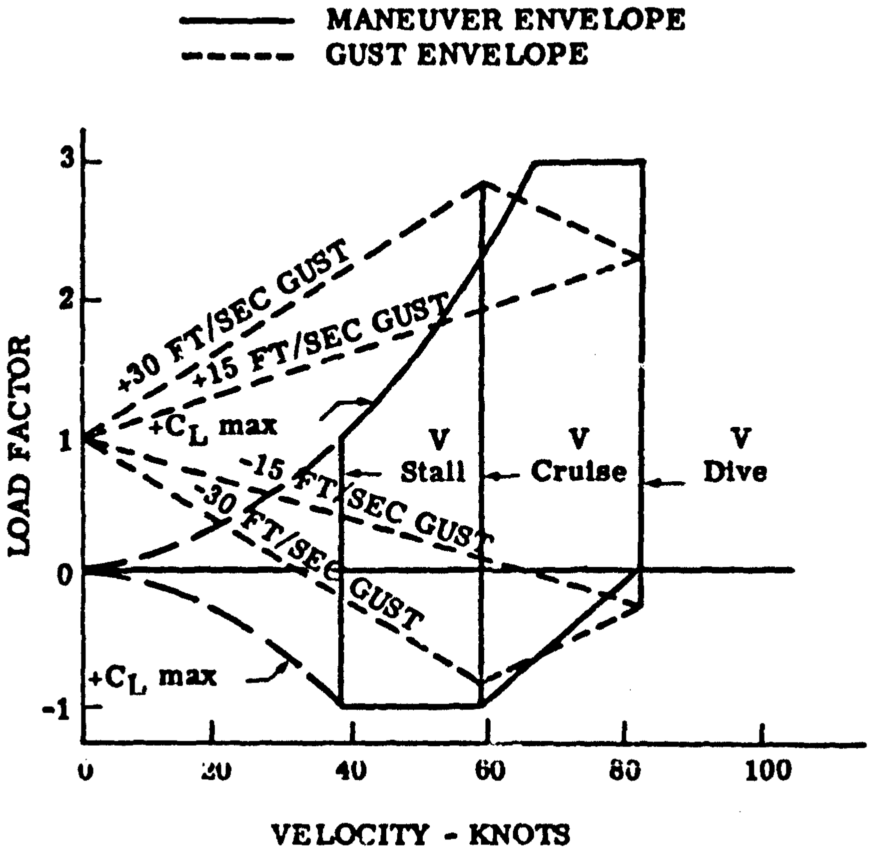
a. V-n Diagram: Figure 8 shows the V-n diagram for the configuration for a 550 pound gross weight. This diagram is based on the requirements of CAR Part 3 with reduced limit maneuvering load factors.

b. Time-Load Study: A brief study of the time-load characteristics resulted in a recommendation that the normal probability curve be used to approximate this date.

c. Tail Normal Forces: A study of the normal forces on the horizontal tail gave a maximum down load of 230 pounds and a maximum up load of 150 pounds.

d. Control Surface Hinge Moments: A brief study of control surface hinge moments gave the following values for maximum conditions as required by C.A.R. 3.

Rudder	H = 40 ft.-lbs.
Aileron	H = 35 ft.-lbs.
Elevator	H = 45 ft.-lbs.



GROSS WEIGHT - 550 LB
WING SPAN - 22 FT

Figure 8. Velocity - Load Diagram

Due to the unknown nature of the structural deflections under airloads there was some question as to what could actually be expected in the way of stability and control. This uncertainty emphasized the importance of full scale wind tunnel tests on the completed airplane.

5. Preliminary Weight and Balance Estimate: The following table represents the weight and balance estimate on the preliminary configuration:

Table I (Sheet 1)
ESTIMATED WEIGHT AND BALANCE

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Wing			38.8	(91.4)		3545	(54.1)		2101
Envelope 23 oz/sq yd		21.9		82.0	1796		56.5	1237	
Ailerons 21 oz/sq yd		11.7		111.0	1299		51.0	597	
Hinges		1.2		106.5	128		52.0	62	
Brace Wires and Patches		1.0		73.0	73		37.0	37	
Air (Δ) at 7 psi		3.0		83.0	249		56.0	168	
Tail 21 oz/sq yd			11.7	(249.4)		2918	(59.9)		701
Vertical		4.5		247.1	1112		72.5	326	
Fin	2.7								
Rudder 1.62# and Hinge .22#	1.8								
Horizontal		6.1		252.0	1537		51.5	314	
Stabilizer	3.2								
Elevator 2.61# and Hinge .29#	2.9								
Brace Wires and Patches		0.8		242.0	194		54.0	43	
Air (Δ) at 7 psi		0.3		248.9	75		60.1	18	
Body			21.6	(82.8)		1788	(36.7)		792
Basic Structure		8.1		147.3	1193		40.1	325	
Envelope 10.75 oz/sq yd	6.9								
Air (Δ) at 7 psi	1.2								
Secondary Structure		13.5		44.1	595		34.6	467	
Canopy 21 oz/sq yd = 2.7#; Windshield = 0.9#	3.6								
Seat & Fairing 21 oz/sq yd	9.5								
Air (Δ) at 7 psi	0.4								

Δ Air at 7 psi was selected as possible maximum.
Total fabric 64.5#; Total air 4.5#.

Table I (Sheet 2)
ESTIMATED WEIGHT AND BALANCE

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Alighting Gear			11.2	(91.0)		1019	(33.4)		262
Main		9.7		66.1	641		21.4	208	
Wheel, Tire, Tube, Air & Axle	6.1			65.6	(400)		19.0	(116)	
Shock Structure	3.6			67.0	(241)		25.5	(92)	
Tail Skid		1.5		252.0	378		36.0	54	
Surface Controls			8.8	(25.1)		221	(31.0)		273
Stick and Supports		4.5		29.3	132		30.0	135	
Pedals and Supports		3.5		10.0	35		29.0	102	
Cables and Fittings		0.8		68.0	54		45.0	36	
Engine Section (Nacelle)			14.4	(110.6)		1593	(71.2)		1025
Engine Mount		12.4		110.7	1373		69.3	859	
Cowling		2.0		110.0	220		83.0	166	
Propulsion Group			63.5	(101.2)		6425	75.8		4813
Engine Installation		46.5		(109.5)	5094		(82.9)	3857	
Engine	38.0			109.3	(4153)		83.0	(3154)	
Carburetor & Air Filter	4.0			103.5	(434)		89.0	(356)	
Exhaust Stacks	1.5			110.0	(165)		79.0	(119)	
Coil (Ignition)	3.0			114.0	(342)		76.0	(228)	
Engine Controls (Throttle, Etc.)		4.0		50.0	200		45.0	180	
Fuel System		7.0		(74.6)	522		(39.7)	278	
Tank	6.0			72.0	(432)		38.0	(228)	
Plumbing	1.0			90.0	(90)		50.0	(50)	

L. E. Wing at Horizontal Distance 59.1.

Table I (Sheet 3)
ESTIMATED WEIGHT AND BALANCE

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Propeller Installation		6.0		(101.5)	609		(83.0)	498	
Propeller	4.0			101.2	(405)		83.0	(332)	
Flange	2.0			102.2	(204)		83.0	(166)	
Pneumatic Group			12.0	(115.0)		1380	(77.5)		930
Compressor (Without Oil Separator)		10.0		120.0	1200		83.0	830	
Plumbing		2.0		90.0	180		50.0	100	
Electrical Group (Battery and Wire)			400	121.0		484	88.0		352
Weight Empty			186.0	(104.2)		19373	(60.5)		11249
Pilot			200.0	42.0		8400	39.0		7800
Gross - No Fuel 21.5% MAC			386.0	(72.0)		27773	(49.3)		19049
Fuel 20 Gal at 6.5#/Gal			130.0	72.0		9360	36.0		4680
Gross - Full Fuel 21.5% MAC			516.0	(72.0)		37133	(46.0)		23729

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Section 2

PHASE II

DETAILED DESIGN

AND FABRICATION

GER 8146

Bruce K. Bain
J.T. Blair

CONFIDENTIAL

PNEUMATIC COMPONENTS

Detailed design of the pneumatic components consisted of laying out the fabric patterns to make up the various parts. Details concerning seam construction, valve locations and methods of attachment of the various sub assemblies were worked out and may be found on the pertinent drawings. After finalization the drawings were sent to Goodyear Tire and Rubber Company for fabrication. A close check was kept on the weight of each of the pneumatic components to insure the lightest possible structure. Shown below are the actual weights of the completed pneumatic components.

Fuselage	7.82 lbs
Fuel Cell	2.98 lbs
Cockpit	13.61 lbs
Canopy	5.78 lbs
Wing (Main section)	26.0 lbs
Ailerons	2.8 lbs
Flaps	3.2 lbs
Horizontal Stabilizer	3.0 lbs
Elevator	2.4 lbs
Vertical Stabilizer	3.1 lbs
Rudder	1.6 lbs

Figure 9 shows several of the completed pneumatic components.

CONTROLS

In the design of the controls emphasis was placed on keeping the size and number of the rigid parts to a minimum for packaging purposes and lightness in weight. Sizes and locations of the control horns were determined from the air loads determined in the section of this engineering report entitled Preliminary Performance, Stability

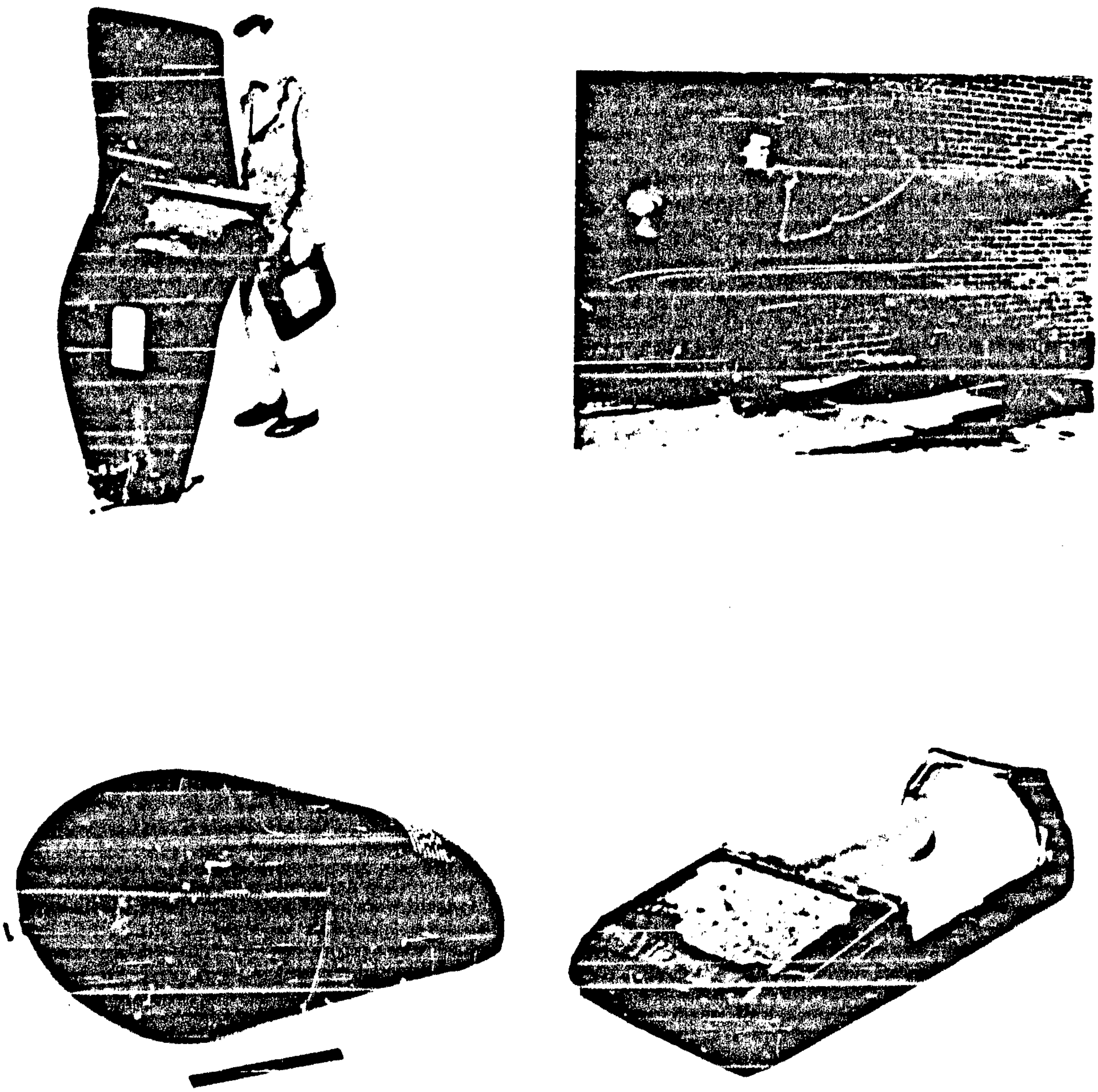

Figure 9. Pneumatic Components

and Control and the control stick dimensions. The control stick was attached to the floor of the cockpit through a larger tire valve base. The base provided a flexible support for the control stock in the bottom pad of the cockpit. The rudder pedals are completely non-rigid and consisted of fabric shoes cemented at the heel to the bottom of the cockpit. The pilot's shoe provides the stiffness to the rudder pedal, thereby eliminating another rigid part in the packaged airplane. The control horns were hinged at the base to lie flat in packaging. The importance of this feature was noticed in packaging the Goodyear Aircraft Inflatoplane. To eliminate the need for a bellcrank and push rod control on the ailerons and yet maintain differential control, standard control cables were used to the under side of the ailerons and bungee chords were used to supply the upward movement to the ailerons. Although the bungee chords put a continuous load into the controls the simplification was well worthwhile. Engine and compressor controls consisted of standard push-pull cables from the carburetor and compressor to the base of the engine mount and cable return systems from these to the cockpit. Teflon tubes were used as cable guides, even for 90° bends. Some of these guides were later replaced by pulleys.

LANDING GEAR

Analysis of the landing loads showed that a 6 inch deflection of the unicycle gear was necessary to absorb these loads. It was found impractical to use fiberglas construction and a tubular aluminum framework was designed. This gear consisted of two concentric rings, the outer one holding the gear in position and the inner one in

and out of the fuselage to absorb some of the landing loads. The unicycle landing gear including the wheel weighed 9.8 pounds (figure 10).

The tricycle landing gear was designed to be easily attached and detached. Shock absorption was accomplished through the use of bungee chords. The completed tricycle gear weighed 28 pounds. The tail and wing tip skids were tubular steel with stellite wearing surfaces attached to laminated fiberglas bases cemented to the fabric.

ENGINE MOUNT AND ACCESSORIES

The locations and mounting details for all engine accessories, routing of fuel, air, and control lines, and details of the mount construction itself were all worked out in the detailed design of the engine mount. All accessories were mounted behind the engine inside a removable cowl to keep drag in this area to a minimum.

It was estimated that under full load the compressor would use 4 hp while 2 hp would be used in turning the compressor against no load. The possibility of putting some type of clutching mechanism into the compressor coupling so that there would be no power drain when the compressor was not being used was studied. The size clutch required was found to be impractical for the power saved and, therefore, was not used. The need for a flexible coupling between the engine and compressor was discussed with Pesco, manufacturer of the pump, and it was decided to omit this item since the pump drive shaft incorporates a flexible coupling to prevent the pump from being damaged by impact loads.

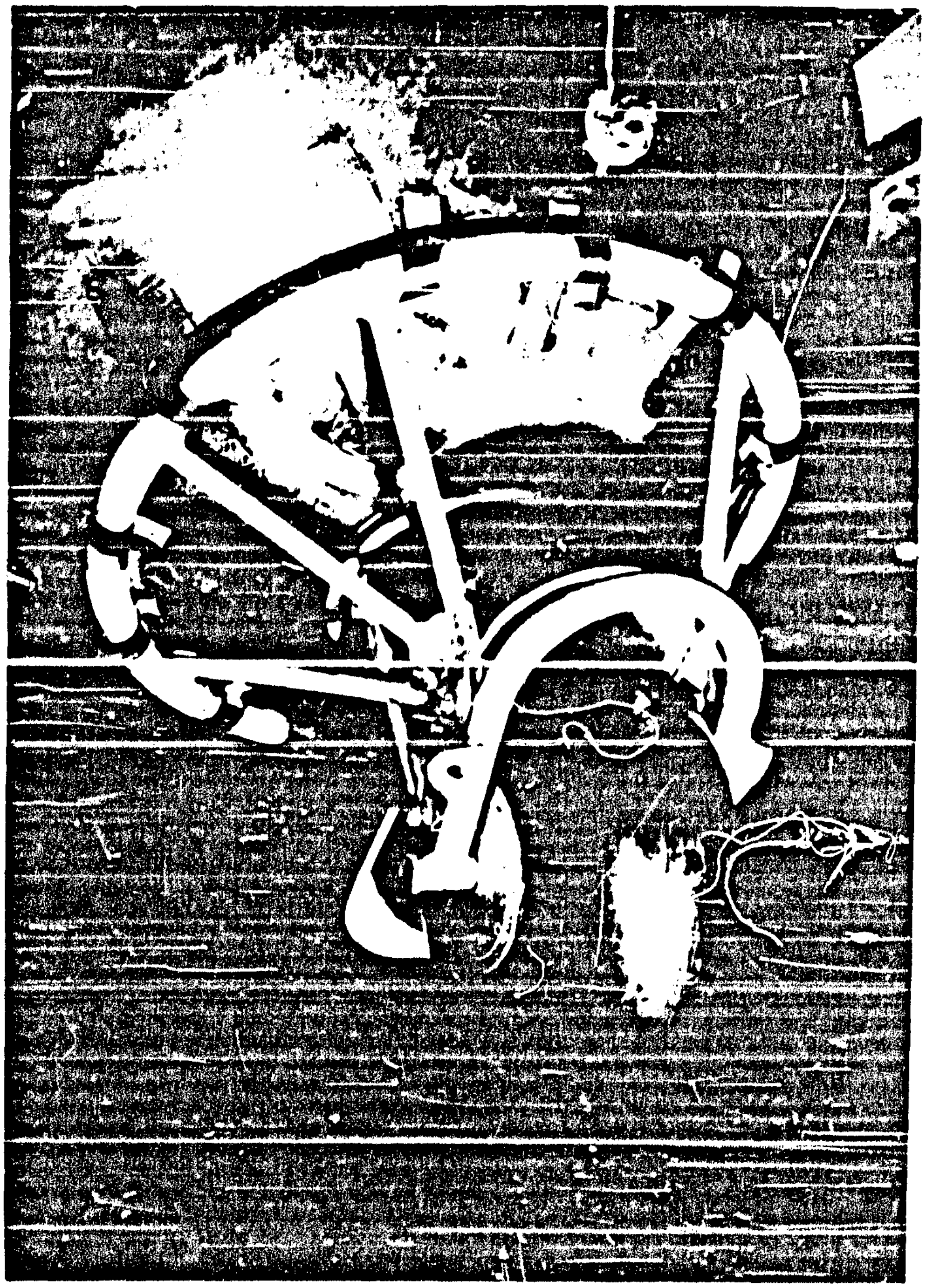

Figure 10. Unicycle Landing Gear

Three different ignition systems were investigated including magneto ignition, water activated batteries, and lead-acid storage batteries. Magneto ignition appeared to be the best answer to the storage problem. The matter of driving both the compressor and a magneto from the rear end of the crankshaft would have required a major redesign of this section of the engine and possible modifications of the crankshaft. However, magneto ignition was felt to be the best solution in the long run. Water activated batteries could be stored indefinitely when kept dry and activated by immersion in fresh or salt water. These batteries were very light, but were a one-shot type and had to be used immediately after activation. Two small lead-acid batteries were selected for use with this airplane since they were readily available, dependable and could be re-used many times with recharging.

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Section 3

PHASE III

PHYSICAL EVALUATION

GER 8146

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Physical evaluation consisted of theoretical analyses and testing to check the load carrying ability of individual components, adequacy of assembly connections, and fatigue on the engine mount and engine operation. An actual weights and break-down was also made. The work accomplished in Phase III is reported in the following sections: Stress Analysis and Static Tests, Engine Tests, and Final Weight and Balance.

STRESS ANALYSIS AND STATIC TESTS

1. Introduction: The Inflatoplane is truly an inflated structure, since only the engine, engine mount, the landing gear (excluding the tire), and sundry cables, connectors, etc., are not made of inflated structure. The wing, cockpit, and tail are made of "Airmat" whose shape is held by means of "drop threads" be-tween two opposite surfaces while the fuselage being a surface of revolution can hold its shape without drop threads. The basic philosophy of design is to main-tain an internal pressure sufficiently large to avoid compressive stresses, and to make the fabric strong and stiff enough to hold the resulting large tensile stresses without failure or excessive deformation, when the design limit loads are applied.

2. Preliminary Tests of Panels: In order to get estimates of the strength and stiffness of "Airmat" materials similar to those used in the Inflatoplane, bending and torsional tests were made with rectangular Airmat panels made of nylon. The

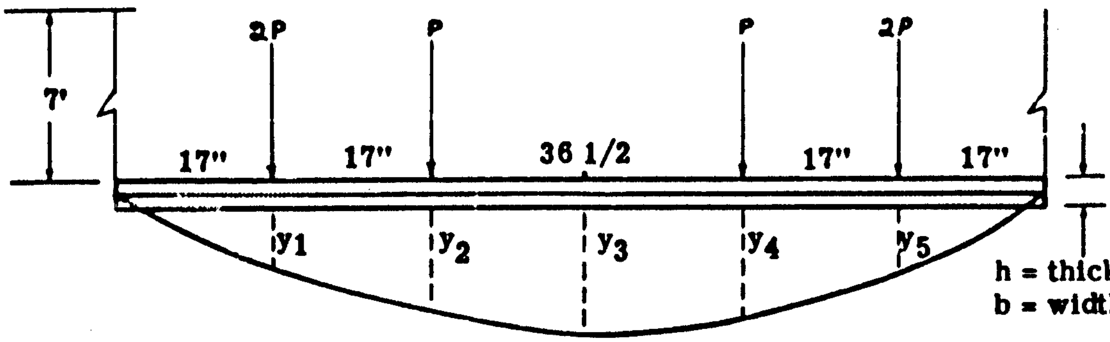
bending tests were made by supporting the panel as a simple beam by means of suspension wires and by applying concentrated loads so that the central third of the panel was subjected to constant bending moment. Loads were applied and subsequently the pressure was decreased until collapse occurred. Deflections and pressures were read simultaneously. The beams were suspended by wires.

The torsion tests consisted of applying various torques to the panel at given pressures. Torques were applied in both directions. A series of tests was made without a bias ply, then the series was repeated with a bias ply added to the same panel.

The dimensions and methods of loading of the bending panels and the panel specifications are shown in figure 11 while the stiffness and strength data are shown in Figures 12, 13, and 14.

The stiffness over weight ratio based on cloth plus cover ply weight per side is 127 and 123 for the 3 and 2 inch panels respectively. Using a stiffness over weight ratio of 125 and 8.125 oz/yards^2 for the wing/side gives E=1000 lbs/in. The final wing had an additional bias cover ply added that increased the weight per side to 9.125 oz/yd/side giving E=1140 lbs/in.

The theoretical curves of Figure 14 are based on the assumption that wrinkling occurs when the compressive stress due to bending is just equal to the tensile stress due to internal pressure. Thus $M_{cr}=P_{cr}\frac{bh^2}{2}$ gives the relation between the critical moment and critical pressure. The data of Figure 14 is inconclusive since the 3 inch Airmat data gives unconservative results below 11 psi while the



3" Bending Panel Fabric Code XA28A198 Wt. 54.1 oz/yd^2 cloth - 4 oz/yd^2/ side cover ply - 2.05 oz/ yd^2/side

2" Bending Panel Fabric Code A322 Wt. 44.05 oz/yd^2 cloth - 3.57 oz/yd^2 side cover ply - 1.0 oz/yd^2/ side

Fig. 11 Bending Panels - Loading & Specifications

2 inch airmat gives conservative results. However data from other sources (reference 8) shows that the theory is conservative at low pressures.

The torsional stiffness data is plotted in Figures 15 - 20. The latter two give the value of G, the modulus of rigidity, as a function of the torque and pressure. All five of these figures show the great importance of the bias ply. From Figure 20 a value of G= 220 lbs/in. can be obtained by interpolation for a pressure of

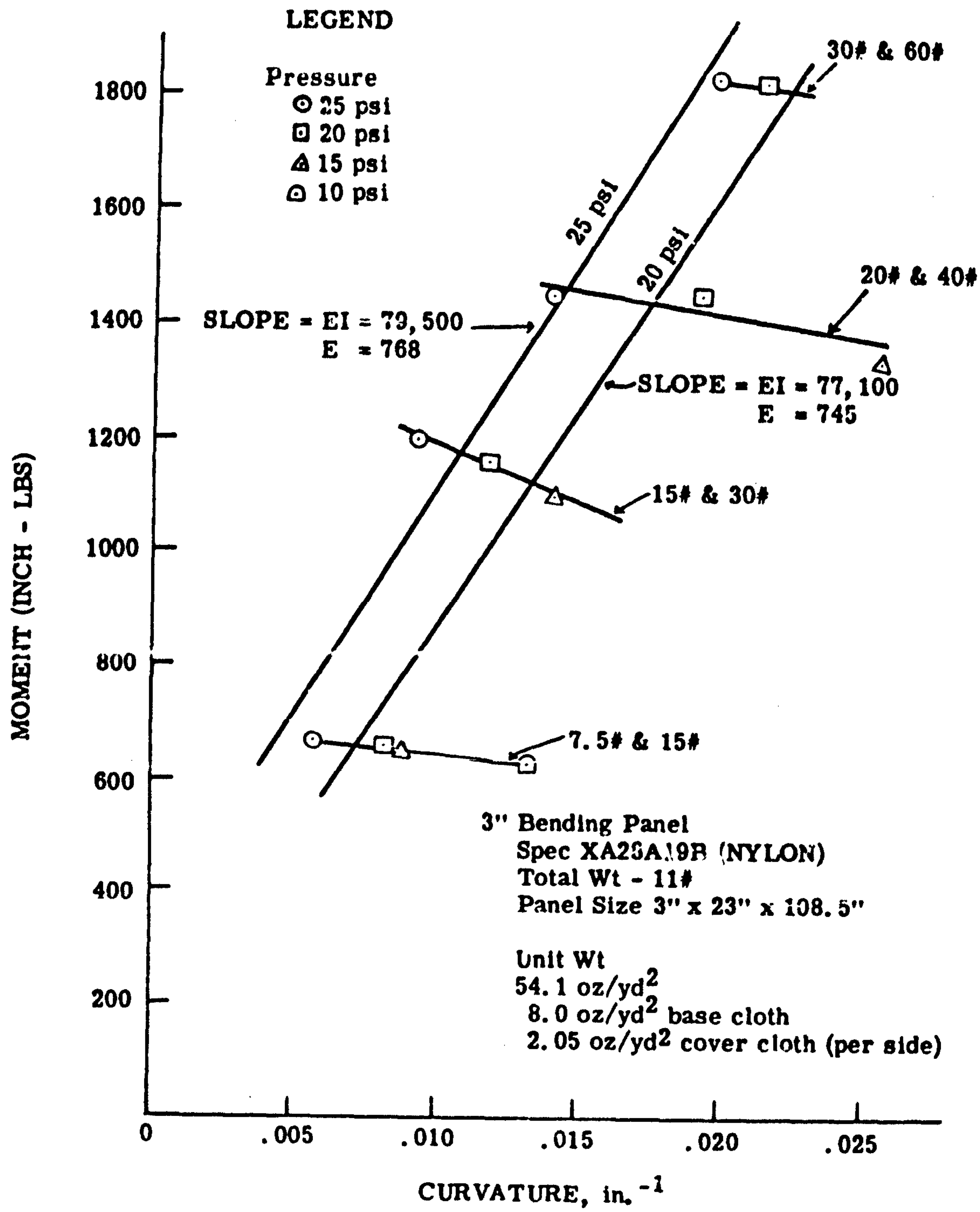


Figure 12. Bending Stiffness of 3-Inch Nylon Airmat Panels

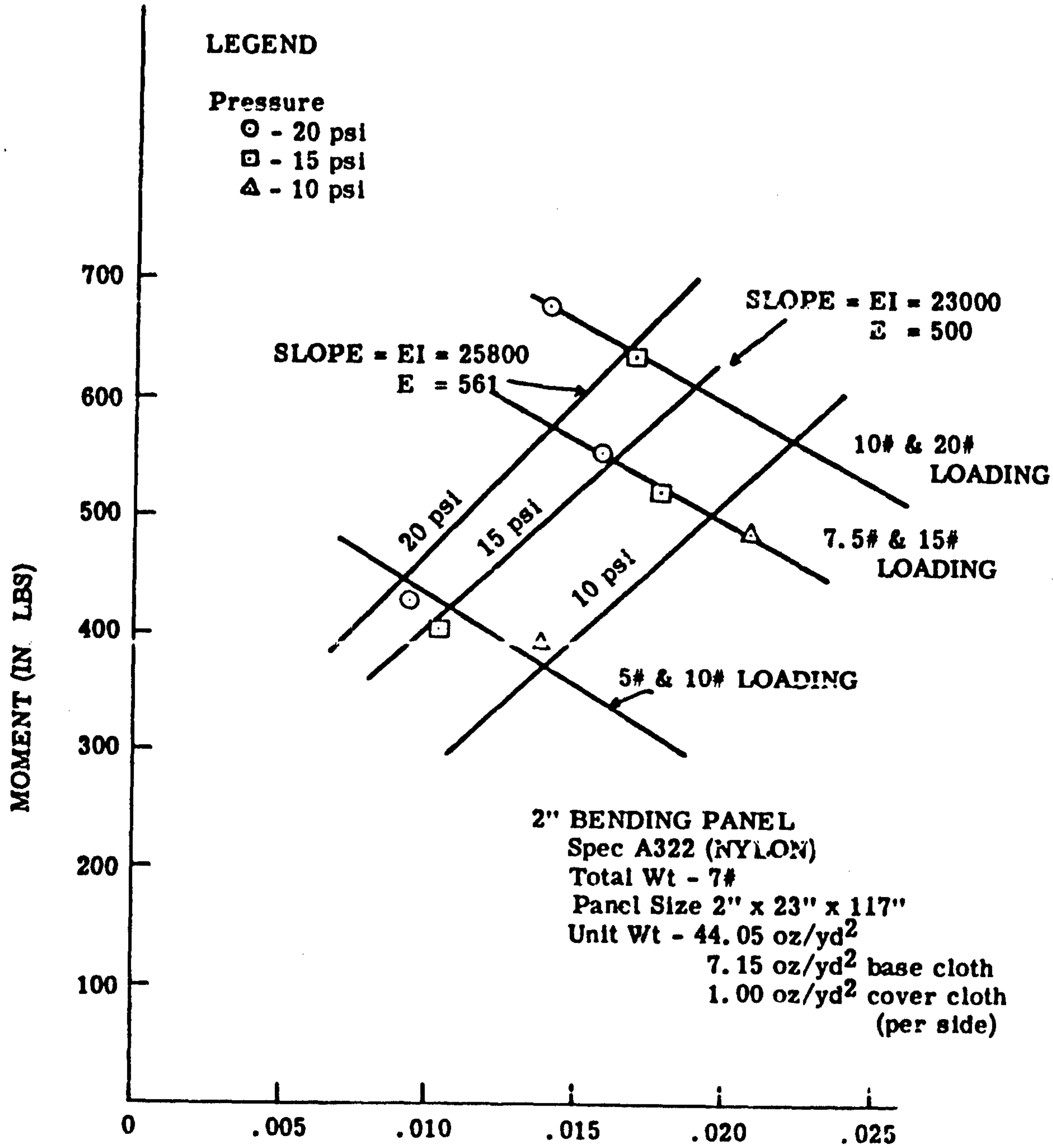


Figure 13. Bending Stiffness of 2-Inch Nylon Airmat Panels

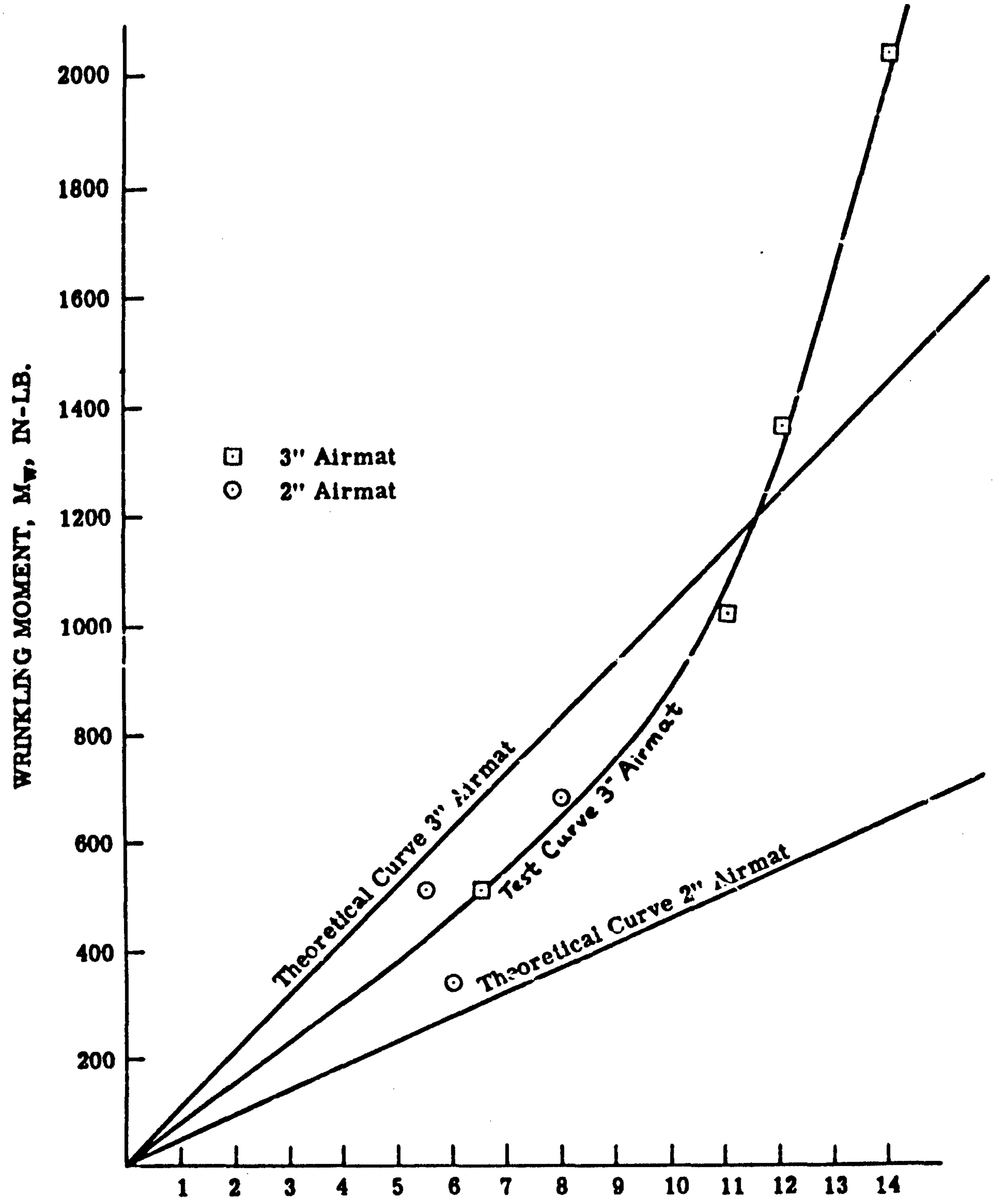


Figure 14. Wrinkling Moment vs Pressure for A 2" and 3" Airmat Panel

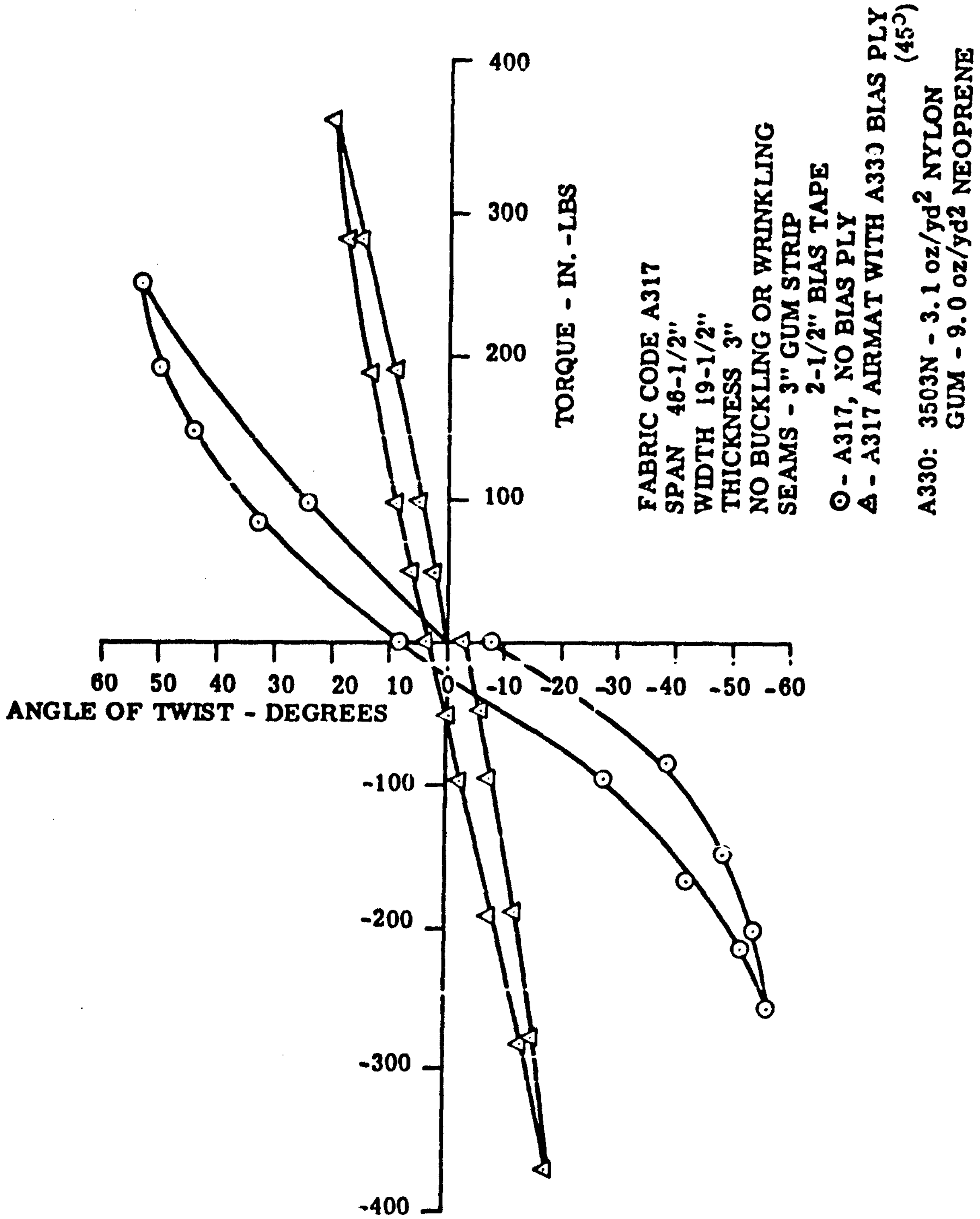


Figure 15. Angle of Twist vs Torque for a 3" Airmat Panel at 2 Psi

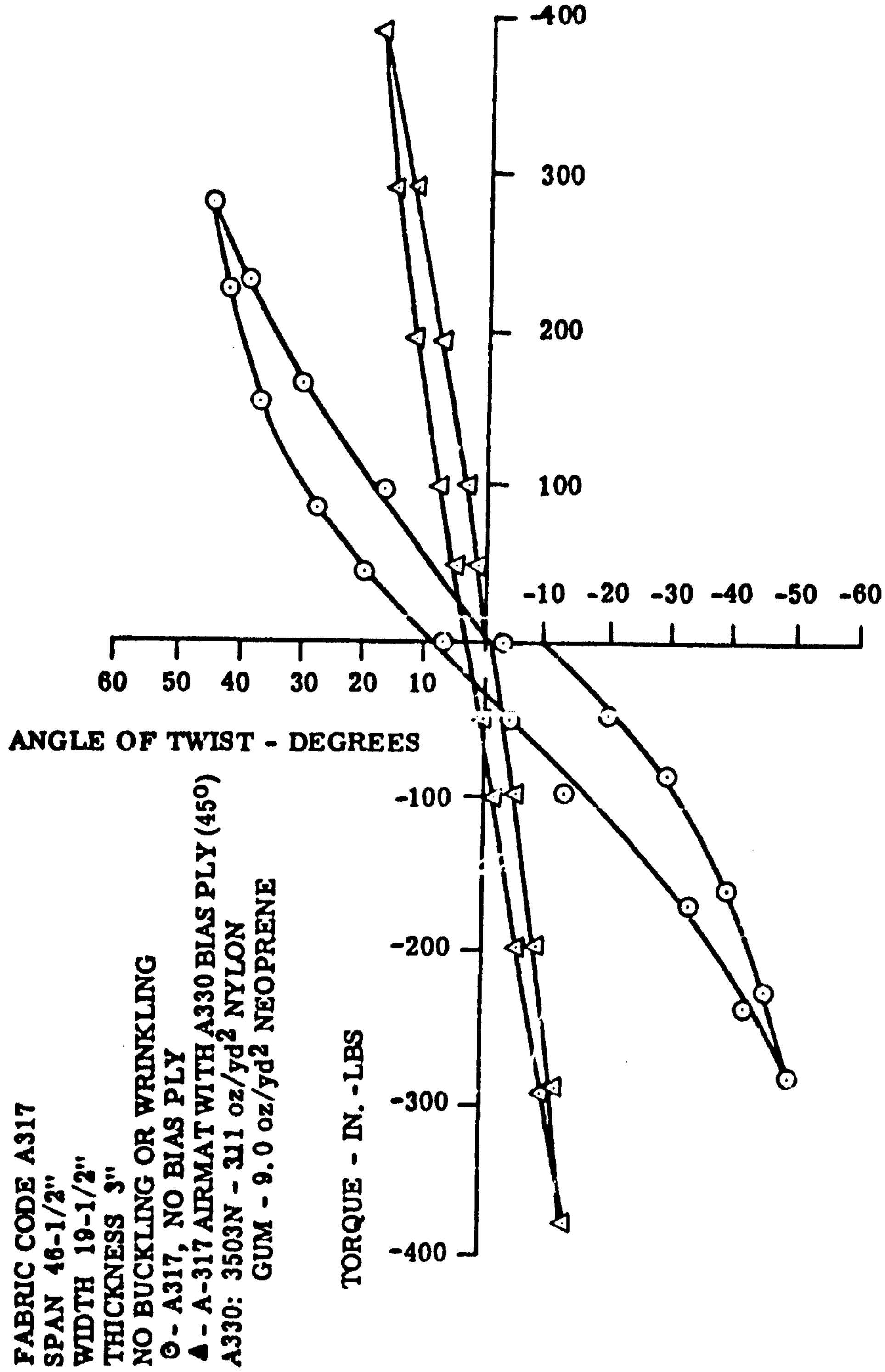


Figure 16. Angle of Twist vs Torque for a 3" Airmat Panel at 5.25 Psi

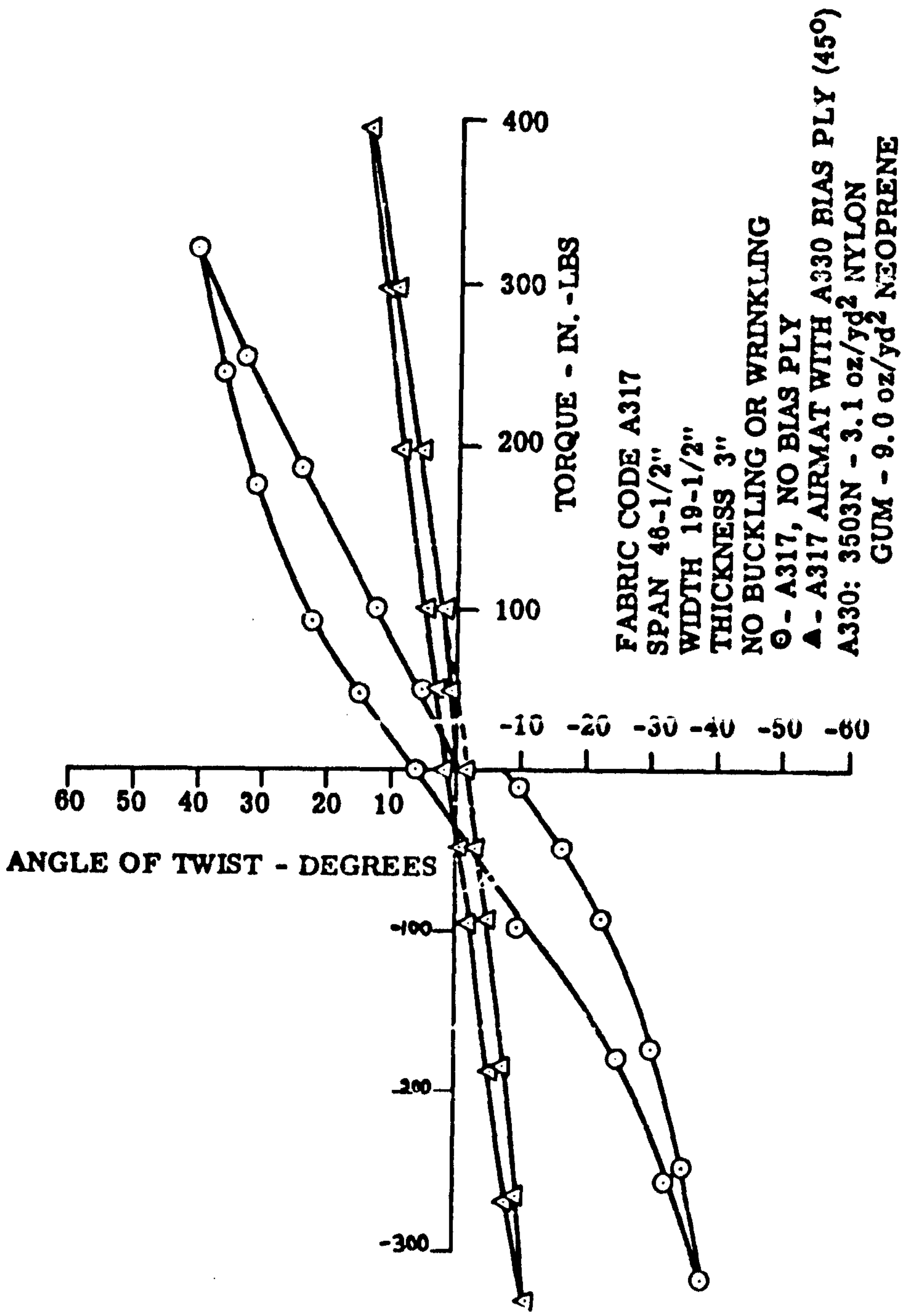


Figure 17. Angle of Twist vs Torque for a 3" Airmat Panel at 10 Psi

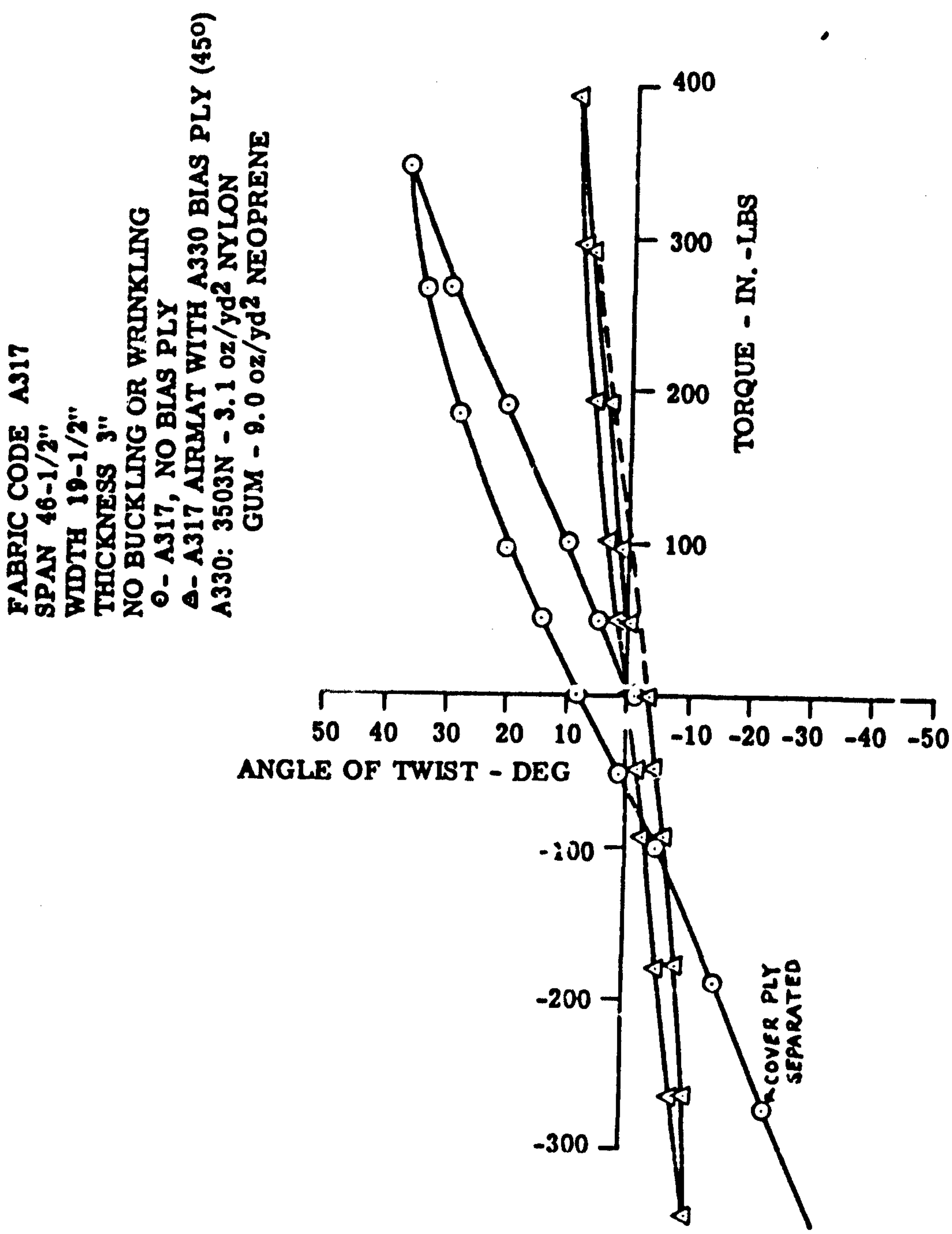


Figure 18. Angle of Twist vs Torque for a 3" Airmat Panel at 15 Psi

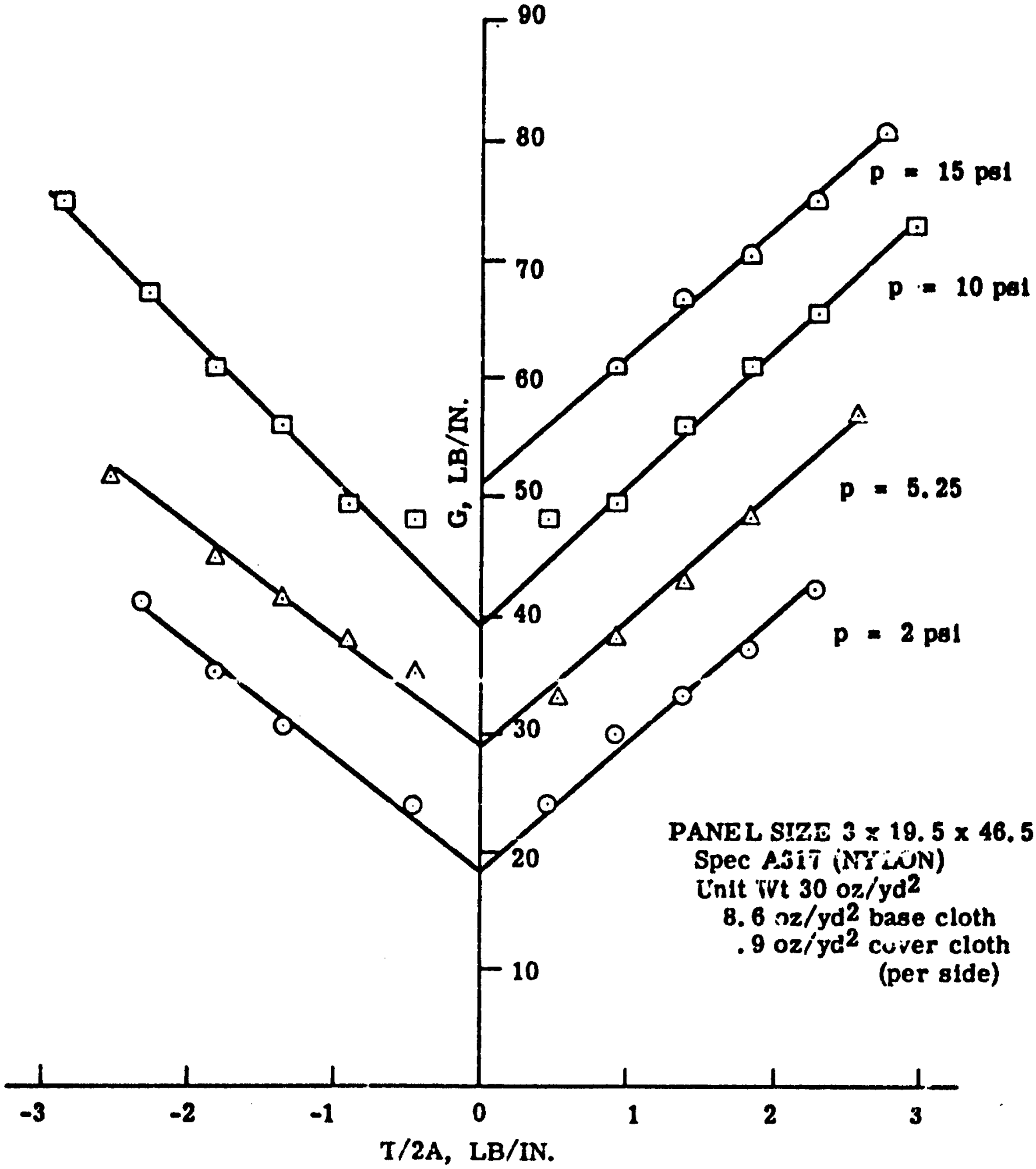
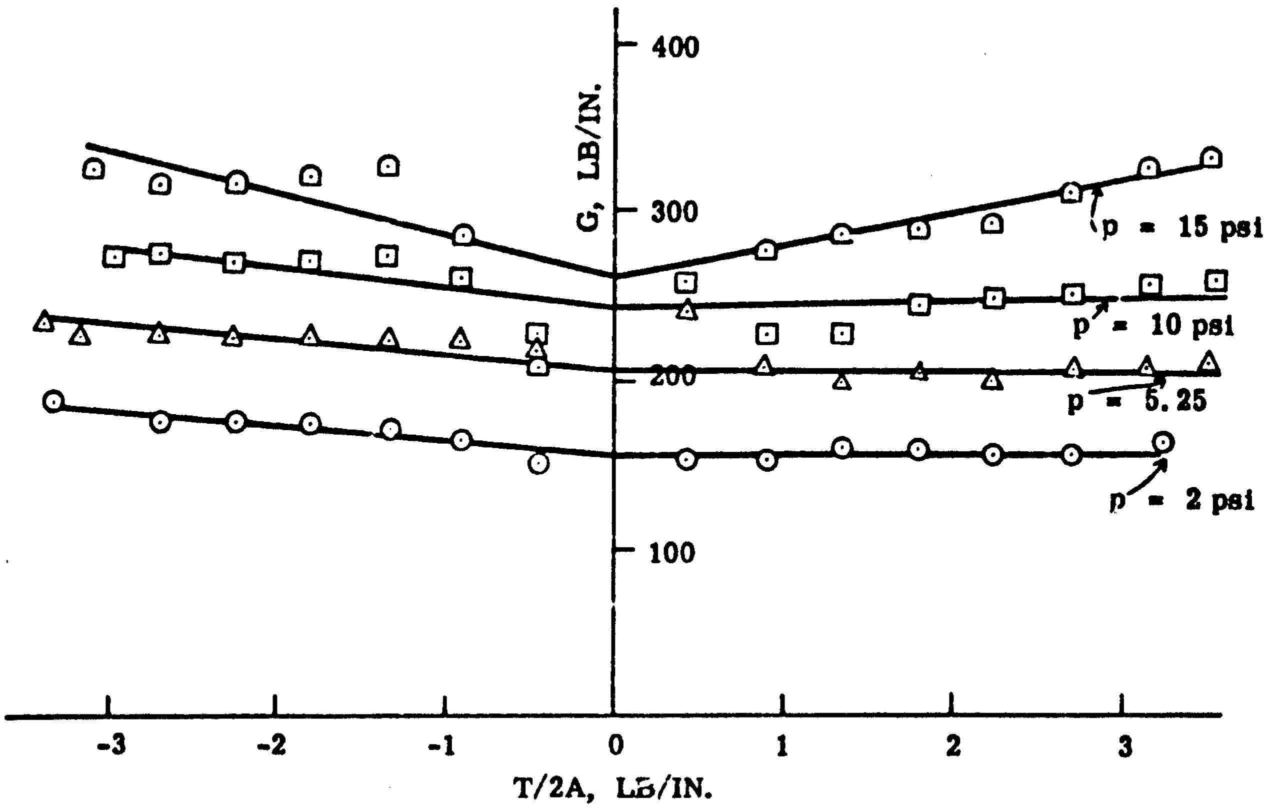


Figure 19. Torsional Stiffness of a 3-Inch Nylon Airmat without a Bias Ply



TORSION PANEL TEST #2

Unit Wt 54.2: Panel Size 3" x 19.5" x 46.5"
Spec A317 (NYLON)
Unit Wt 30.0 oz/yd^2
8.6 oz/yd^2 base cloth (3514N)
0.9 oz/yd^2 cover cloth (per side)
Spec A330 (BIAS PLY) 45°
3.1 oz/yd^2 (3503N NYLON)
9.0 oz/yd^2 (GUM - NEOPRENE)

Figure 20. Torsional Stiffness of a 3-Inch Nylon Airmat with a Bias Ply

7 psi and T/2A = 2. This value can be increased to 240 lbs/in. by using the weight ratio of the wing and test panel cloth plus cover bias plies.

3. Analysis of Inflatoplane:

a. Wing: Because of the relatively low stiffness of "Airmat" and strength at the pressures used, it was necessary to use cables to support the wing. Each half span was supported by two cables from above and two cables from below. The cables were installed in pairs for rigging purposes and to decrease torsional deflections. The lower cables were fastened to the landing gear and the upper cables were fastened to the engine mount.

The cable loads were determined by assuming no motion of that section of the wing to which the cables were attached. This assumption seemed reasonable since a load of 10 lbs applied 76 inches outboard of the center-line (the point where the cables were attached) would deflect the wing 1-inch. The original calculations were made by assuming that the airload was uniformly distributed spanwise whereas the actual airload has an elliptical distribution spanwise. Thus while the cable attachment points are not located ideally for either a uniform or elliptical distribution the locations chosen were acceptable for either.

The computations showed that the wing would take a little more than 3 g's upward load (1 g = 550 lbs) and 1 g downward load. These two are different because the upper wires were fastened to the engine mount at a point behind the wing thus causing increased torsional load and unsymmetrical bending.

b. Tail: The tail was designed using assumptions similar to those for the wing. Cables were located at the hinge line to minimize torsion and at the leading edge to minimize flutter. The elevator hinge line cables were located at the outboard end of the horizontal stabilizer because it is continuous at the fuselage, while the rudder hinge line cables were located below the tip of the vertical fin since it was hinged at the fuselage. The compressive loads caused by the cables were small compared with those caused by the wing cables.

c. Fuselage: The fuselage was treated as an inflated cone subjected to bending loads from the tail. To avoid excessive deflections it was found necessary to aid the fuselage by supporting some of the tail load with a cable attachment to the engine mount.

The engine loads were transferred to the fuselage by means of a belly band. The tricycle gear landing loads were transferred to the fuselage through pads on each side and by means of a belly band. The shock loads from the unicycle landing gear are transferred into the fuselage by a piston-like device held in position by a ring fastened to the forward hemispherical part of the fuselage.

d. Cockpit: The cockpit was considered to be a channel whose 'flanges' and 'web' were made of "Airmat" panels. The side panels (flanges) were fastened to the sides of the fuselage and to a bulkhead at the hemispherical nose of the fuselage. Straps fastened on each side of the pilot's seat and along the top of the fuselage helped support the cockpit. Critical bending moments were

calculated to occur just forward of the strap attachment points under a positive 3 g loading condition. Negative margins for a wrinkling failure were calculated for several conditions. Possibly the theory is inadequate with structures of unusual cross-section such as the cockpit, since no difficulties with the cockpit were encountered in flight testing.

e. Engine Mount: The engine mount consisted of a saddle laced to the wing and fastened to the fuselage by means of a belly band, and a triangular frame hinged to the saddle and belly band assembly. The triangular frame itself was three times redundant while an additional redundancy resulted between the frame and the saddle. It was also necessary to assume that the pressure between the saddle and wing was trapezoidally distributed. These redundancies made it necessary to separate the engine mount assembly into free bodies at hinge lines and to determine the hinge reactions and internal bending moments by minimizing the elastic energy of the engine mount assembly. The loads on the engine mount consist of the thrust and torque of the engine, inertia loads from the engine, batteries, compressor, and the cable loads from the wing and tail. The engine mount had already been designed, analyzed, and built before the wing and tail cables were attached to it. When these were attached, the critical loading condition was that due to a negative lg load on the wing. This loading gave negative margins in the forward oval tube and the lower circular tube of the triangular frame.

f. Landing Gear:

1) Unicycle Gear: A piston-like device pressing into the hemispherical end of the fuselage absorbed the shock load of landing. A ring and truss supported the piston and the fore and aft components and side components of the landing loads. The energy absorption capacity of the piston device would not exceed that due to a descending velocity of more than 5 ft/sec. With this descending velocity the single wheel gear was adequate for all the other loading conditions in the Civil Aeronautics Manual 3.

2) Tricycle Gear: In order to withstand a descending velocity of 7 ft/sec. as required in the Civil Aeronautics Manual 3 and for training purposes a tricycle gear was designed. The shock load was absorbed by bungee cords fastened by means of pads and a belly band to the fuselage. The fore and aft and side loads were supported in part by the unicycle assembly.

4. Test Results:

a. Wing: The wing was tested by applying a series of 25-pound shot bags along the quarter chord line. In order to simulate the elliptical spanwise load distribution the shot bags were placed at varying distances from the wing center-line as shown in figure 21.

Preliminary tests were made using a uniform distribution. The results were unfavorable so the elliptical load distribution was used. Two tests were run with the elliptical distribution and the plane in the inverted position (figure 22).

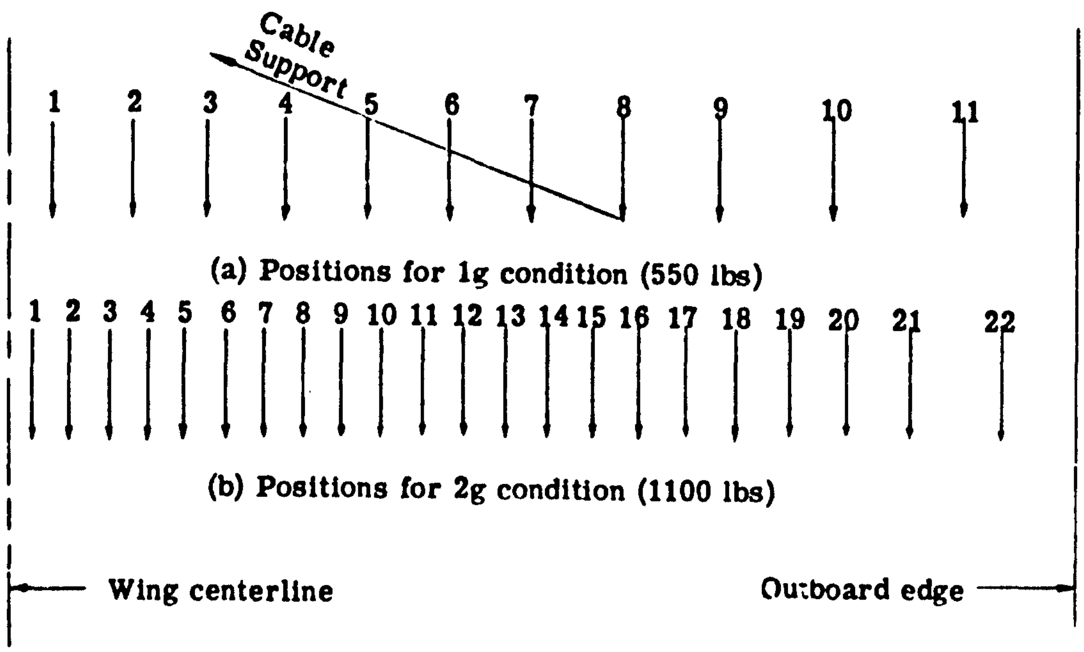


Fig. 21. Relative Positions of 25 lb. Shot Bags for Static Test of Wing (one half span).

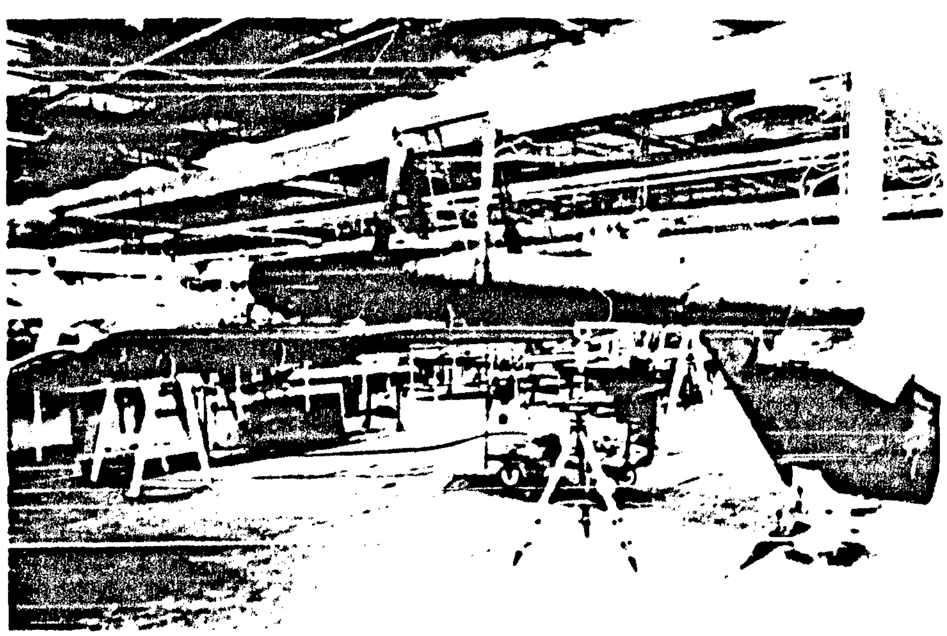


Figure 22. Wing Static Test with 1g Up-Load

In Test I, the load system shown in Figure 21a was used with the bags being applied in ascending numerical order.

This lg load was supported successfully. In order to get a two g condition another figure 21a loading system was superimposed on the existing figure 21a loading system. Failure occured by buckles at the leading edge of wing where it was attached to the bulkhead at the forward end of the fuselage.

In Test II, the load system of figure 21b was used, the bags being applied in ascending order as numbered. This loading system caused shear buckles to appear at the leading edge of the wing near the bulkhead. However, the wing did support this load. The test was continued by applying loads 1 - 5 of figure 21a in ascending numerical order. Load No. 5 caused complete collapse of the wing by a compression buckle appearing half way between the fuselage and the cable attachment points. The last five bags were removed and the wing was aided manually until it could support the 2g load system of figure 21b, unaided. Loads 2, 4, and 6 of figure 7a were then applied in that order. When load No. 6 was applied complete collapse occurred as in the preceeding portion of Test II.

The wing successfully supported the negative lg load system of figure 21a. After this test was performed a crude test was made to determine the stiffness of the wing. Two 25 pound loads were applied, one at the shear center of each outboard edge. Deflections were measured at the cable attachment points, the outboard edges, and midway between these points. These measurements gave

an EI = 1.2×10^6 lbs/in. for the wing. An EI = 1.5×10^6 lbs/in. was predicted by the panel tests. In order to appreciate this stiffness value it should be noted that the point of attachment of the cable to the wing deflected about 8 inches due to the negative lg loading condition. The outboard edge deflected 3.8 inches due to one 25-pound load applied there.

After observing the shear buckles in the wing leading edge computations were made to determine the reason for their appearance. These computations showed that shear buckles would occur and form a tension field in the leading edge at about a 2g load. The shear buckles plus the uncertain relocation of the fuselage point of attachment of the cable support, contributed to the collapse of the wing at a little more than 2g. It should also be pointed out that when inflated the wing proved to be longer, narrower, and thinner than designed, all of which made the wing weaker. The shrinkage in the wing cloth in curing caused a 12 percent reduction in cross-sectional area. These dimensional changes were caused by shrinkage in the wing material in processing.

b. Fuselage and Tail: Proof loads were applied to the tail to test both the tail and fuselage. These loads showed that the tail was satisfactory and that a cable support to the engine mount was necessary for the fuselage.

c. Cockpit: A 3g load applied to the cockpit caused a small buckle to appear where the wing cables pressed into the bottom panel (figure 23). The critical section of the cockpit just forward of the pilot's seat was not tested since the load was placed behind this point.

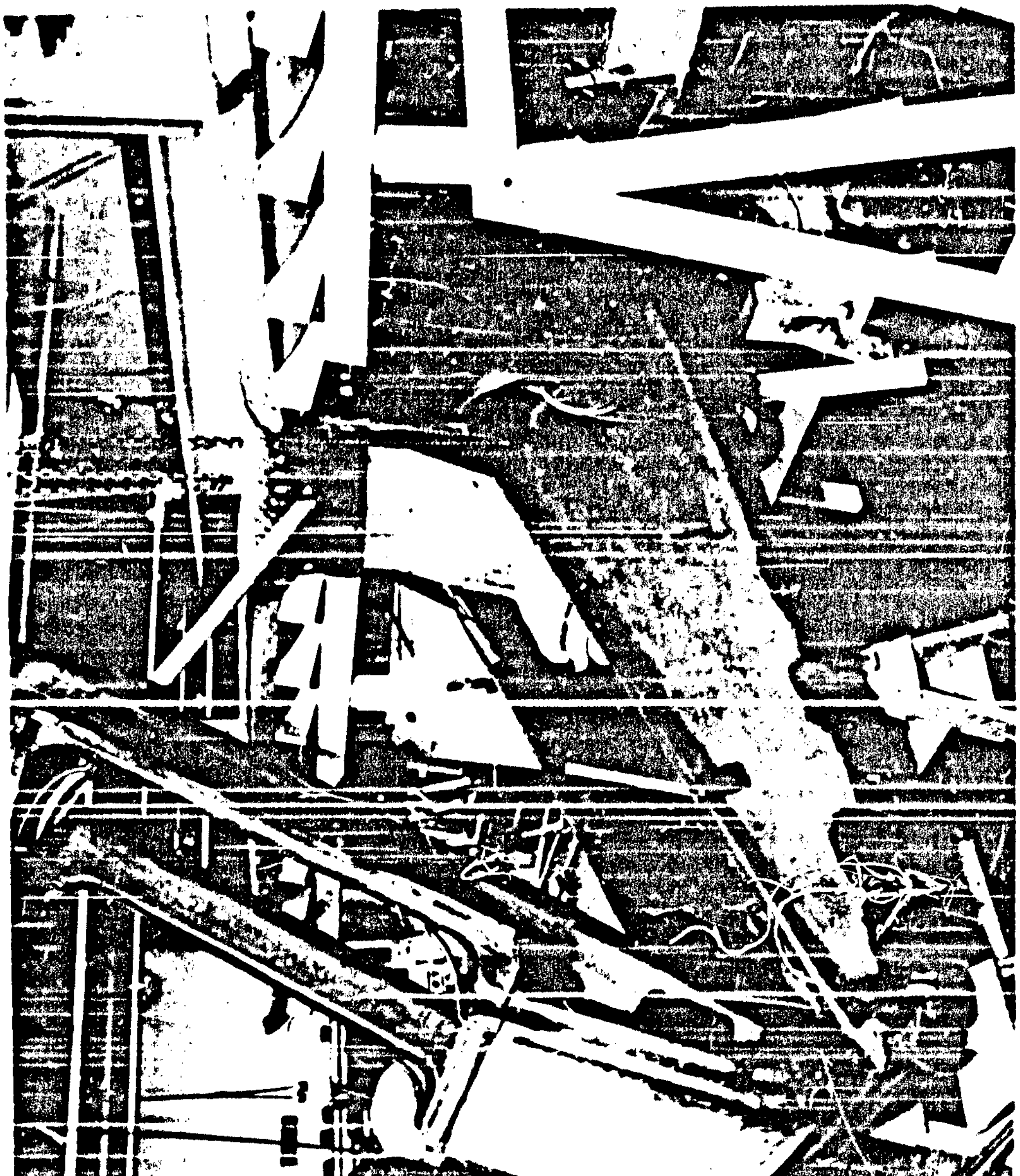


Figure 23. Cockpit Static Test with 550-Pound Load

d. Engine Mount: Since negative margins were calculated in the vertical oval tube of the engine mount frame with the negative lg loads condition it was necessary to test the engine mount for this condition. This was done by first applying the negative lg loads to the wing, and after removal of this load, the engine thrust but not the torque was applied. Stresses were measured by SR-4 strain gages mounted on the oval tube, at points on the saddle, and on the cables supporting the wing and tail. The stresses for the two loads were then added.

These measurements showed that this condition was not critical for the engine mount because the cables from the wing had only about half the calculated load in them and that the load distribution of the saddle reaction was not as assumed. Even though no strain gages were placed on the aft streamline tube bending stresses could be calculated for this tube from the bending stresses in the oval tube and the known thrust and cable loads. The bending stresses in the streamline tube were greater than those calculated and thus showed that more of the load was carried by the aft tube than the theoretical calculations showed.

The small load in the cable wires resulted in large deflection of the wing. There is no doubt that the cable deformations were very small; hence, the wing deflection was caused by yielding supports. Purely geometrical calculations show that the wing could deflect 8 inches if that point on the engine mount to which the cables were attached moved forward 2 inches while the wing compressed 1 inch. Observations made during the test indicate that this combination of yielding supports is not unreasonable. Yielding of supports of course reduces the strength

of the wing.

5. Conclusions and Recommendations: Since the stiffness of "Airmat" is much less than that of other structural materials it is necessary to have more refined methods of measuring both the strength and stiffness. One illustration is the need for taking into account the change in geometry during test. The use of other fabrics such as Dacron would help stiffen the structural components. A pickless fabric in the fuselage would stiffen it.

The monocoque construction of the wing is not effective for carrying large shear stresses. The introduction of shear webs into the wing ought to be investigated. Rather large yielding of the support to which the wing cables are attached ought to be minimized. Adding another set of cables would decrease the shear load in the wing but the yielding of the support for the cables would be even more critical than with only one set of cables. Allowance should be made in the original construction of the wing to insure the desired final dimensions.

The cockpit would be able to carry more load if it were not supported as a cantilever or if the pilot were moved 6 inches aft which, of course, shortens the cantilevered loads on the cockpit. Supporting the cockpit along the sides would also help a great deal in carrying side loads which cause torsion in the present configuration. Such support would result in more favorable shear distribution in the wing.

ENGINE TESTS

1. Introduction: The purpose of this section is to present the results of engine operation on the GA-447 Inflatoplane and to comment on the installation features of the power-plant assembly. The information presented includes the results of the engine manufacturer's static tests and Goodyear Aircraft's static and flight tests. Included as Appendix B of this report is a discussion of carburetor icing in regard to the installation.

2. Description of Installation:

a. Engine: The Nelson H-59A engine, manufactured by Baromotive Products, Incorporated, San Leandro, California, is an air cooled, four cylinder, horizontally opposed engine operating on the two-cycle principle. A six volt, battery ignition system is used and the engine operates on a fuel-oil mixture in the ratio of eight parts of fuel to one part of oil by volume. The model H-59A engine is an improved version of the model H-59 in that cylinder bases have been modified to provide better cooling. The crankshaft has been redesigned to withstand operating stresses without failure. The basic engine as received from the engine manufacturer has been modified by Goodyear for installation on the GA-447 Inflatoplane to the following extent:

1) The exhaust stacks furnished with the engine were removed and replaced with individual, light weight, short stacks.

2) A ram air scoop was fitted to the carburetor air filter.

3) A Pesco, Model 3P-485, air pump was mounted on the starter housing to

be driven by direct coupling to the engine crankshaft. This modification involved removal of the starter assembly and ignition breaker cam and replacement by an ignition breaker cam drive assembly. The pump drive shaft is connected to the cam drive assembly by a short, internally splined coupling. Since the cable operated starter assembly is removed, engine starting is accomplished by hand "propping."

b. Propeller: A V.S. Propeller, 47-inch diameter, fixed pitch, wood propeller was used. The tips of the propeller have been covered with a plastic coating to protect the thin cross-section.

c. Accessory Section: The accessory section enclosed by cowling consists of the air pump, air pump oil supply tank, the combination air control and check valve, air pressure relief valve, oil separator, batteries and ignition coils.

d. Engine Mounting: The powerplant assembly is mounted on a pedestal type aluminum alloy mount that is installed above the wing on the center-line of the fuselage. Vibration isolators are used between the engine lugs and the brackets on the engine mount. The mount also provides support by brace wires to the wing and empennage of the airplane.

e. Fuel System: The fuel system consists of a bladder type fuel cell that is installed inside the forward section of the fuselage, a fuel pressure regulator and a downdraft, float type carburetor. In operation, fuselage air pressure compresses the fuel cell forcing fuel to the regulator mounted at the fuel inlet of the

carburetor. The regulator reduces the fuel pressure to 1.0 ± 0.5 psig which is normal operating fuel pressure required. The regulator also incorporates a filter element to prevent dirt from entering the carburetor.

f. Electrical System: The electrical system consists of two six volt storage batteries, two ignition coils, an ignition breaker assembly, spark plugs and cockpit ignition switch. This system is used solely to provide ignition of the fuel-air charge. Since no generator is included in the system, engine operation is dependent on battery life.

g. Air Supply System: The air system used to supply air and maintain airplane internal pressure in the event of leakage consists of an engine driven air pump, a combination control and check valve that is operated manually from the cockpit, a pressure relief valve and an oil separator. Since the air pump operates continuously and air flow is required only in the event of leakage, the combination valve is used to control air flow. This valve functions to exhaust the pump output under the normal no-flow condition and upon manual actuation from the cockpit directs the pump output to the airplane envelope. Under the normal no-flow condition it also functions as a check valve to prevent loss of envelope pressure. The pressure relief valve functions to maintain a safe operating pressure in the system and the oil separator removes oil vapor from the air delivered by the pump.

h. Instrumentation: On the basic airplane no provision is made for powerplant

47" Goodyear Propeller Test
Make - US Propeller - Ser #918-380-31
Engine - Nelson H59A 501 #222
Points indicate Power Absorbcd by
Propeller at Various Speeds
Tests conducted 12-17-56 - T. Ne'son

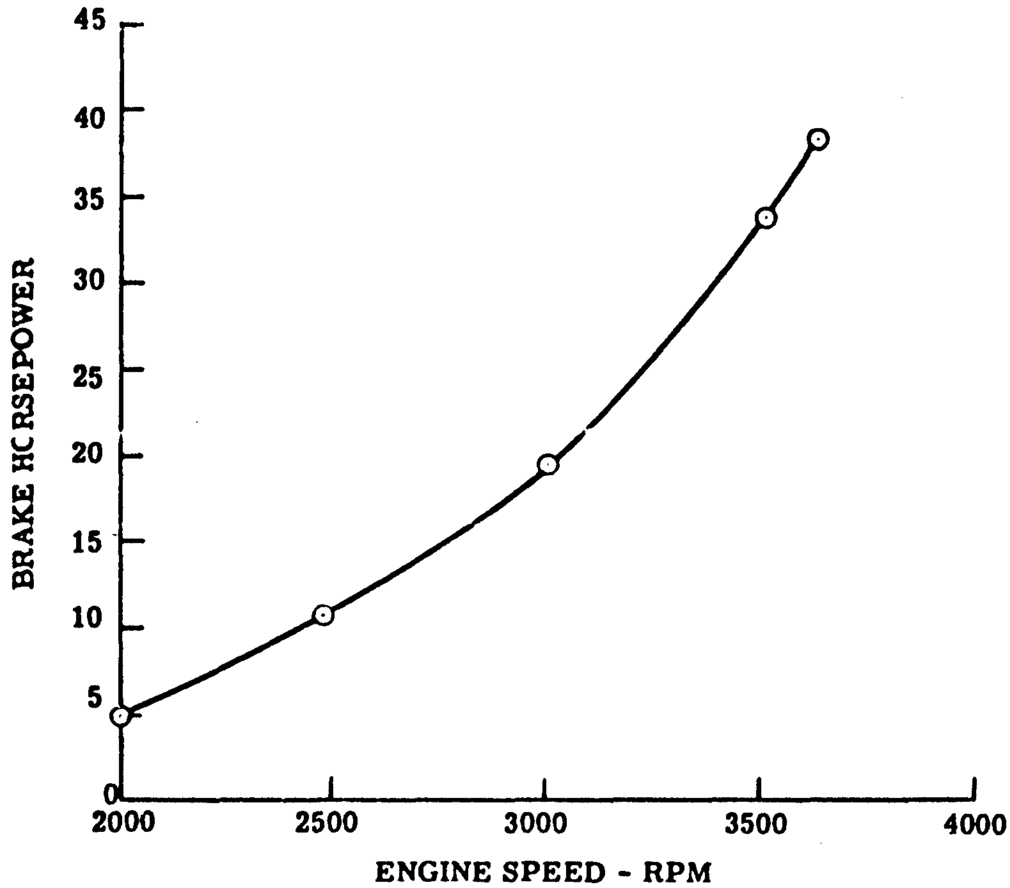


Figure 24. Engine Speed vs Horsepower with 47" Propeller

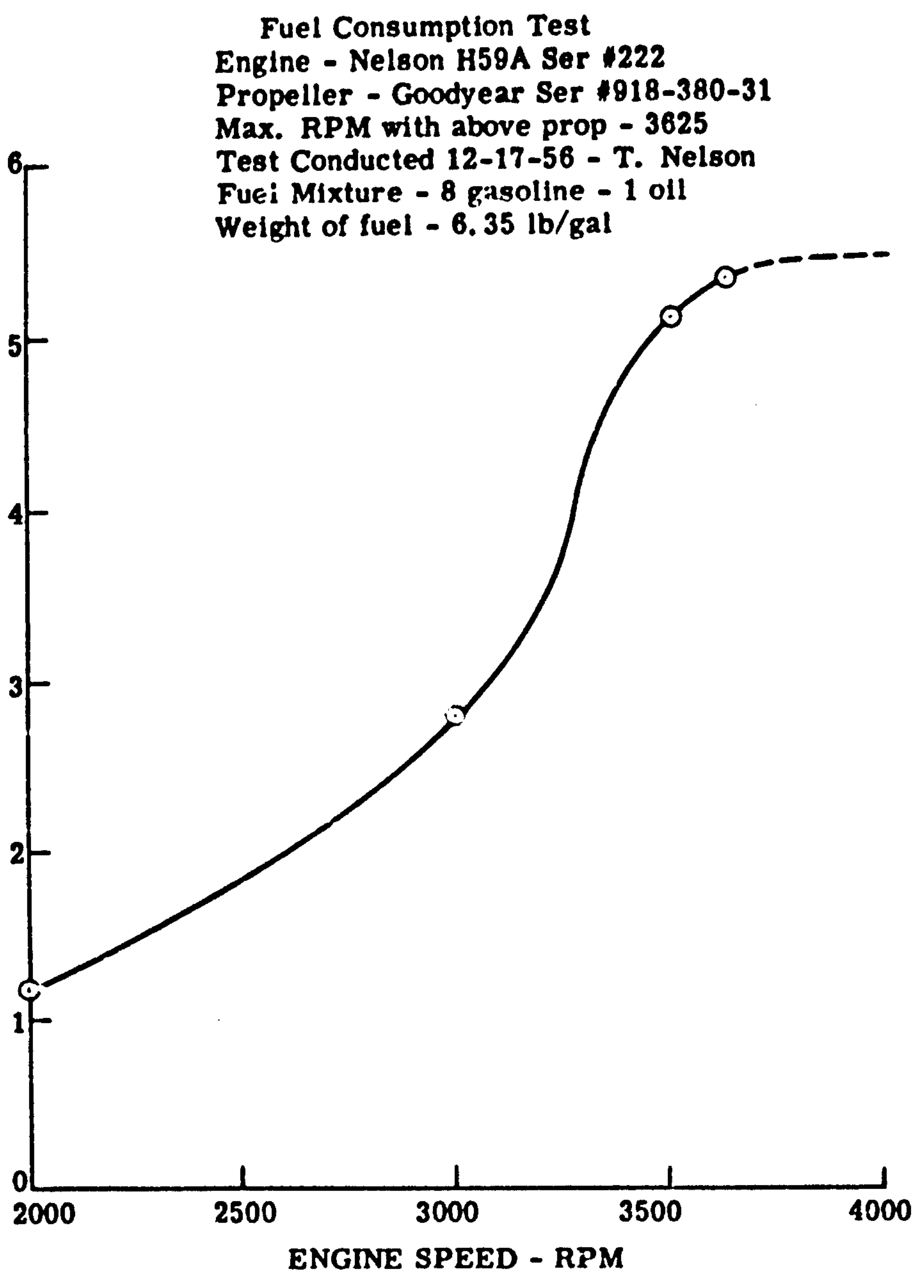


Figure 25. Engine Speed vs Fuel Consumption with 47" Propeller

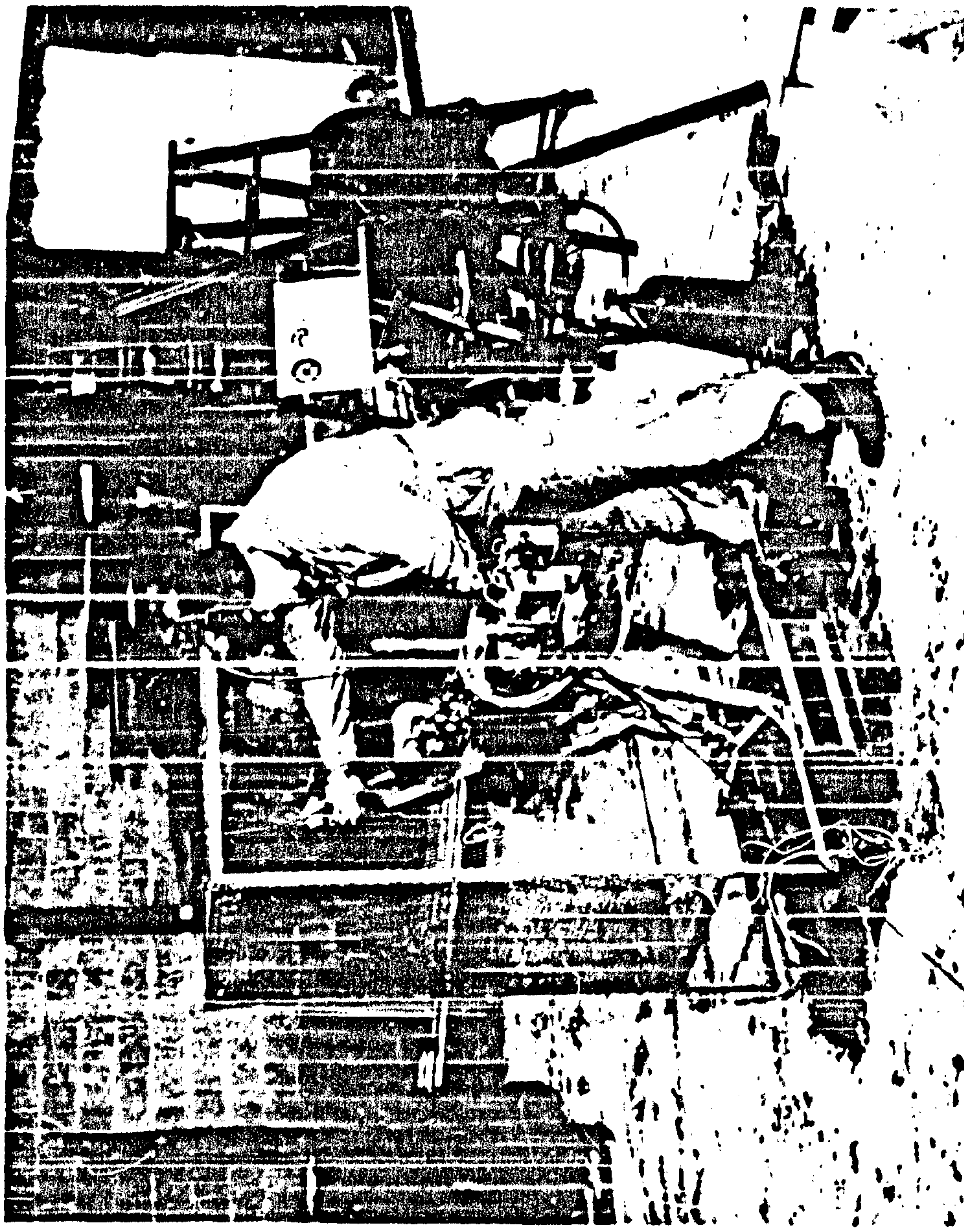


Figure 26. Engine Test Set-Up

instrumentation. However, for static and initial flight operation, thermocouples were installed on both rear cylinders to obtain cylinder head temperatures and an automotive type electrical tachometer was installed in the primary circuit of the ignition system to obtain engine speed measurements.

3. Powerplant Installation and Operation: This section of the report presents the results of static operation of the power-plant installation to determine its operating characteristics. These static tests were conducted by Baromotive Products to determine brake horsepower, fuel consumption and engine cooling characteristics; and by Goodyear Aircraft to determine the effects of engine operation on the pedestal mount structure, the ability of the engine to drive the air pump, and to check the functional operation of the complete installation.

a. Baromotive Products Static Tests: The first test runs made with the engine-propeller combination were conducted after the standard factory break-in runs. A total of four runs was made at different engine speeds to determine brake horsepower, fuel consumption and cylinder head temperatures. The results of these runs and the engine break-in runs are shown in Table II. Maximum engine speed reached with the 47-inch diameter propeller was 3625 rpm and the horsepower obtained at this speed was 38.25. The design of this propeller is such that full power cannot be developed under static conditions. Figure 24 presents a curve of brake horsepower versus engine speed and figure 25 presents a curve of fuel consumption versus engine speed. At rated engine speed fuel consumption is approximately 5.5 gallons per hour. Engine cooling was satisfactory with all

cylinder head temperatures remaining below the maximum recommended limit of 450° F.

b. Goodyear Aircraft Static Tests: A model H-59 engine with a 42-inch diameter propeller was mounted statically to determine the effect of engine operation on the pedestal type engine mount (figure 26). The air pump was mounted on the engine to provide proper weight distribution but was not coupled to the crankshaft since the ability of the model H-59 crankshaft to drive the pump was uncertain.

Table 2 presents the results of the static test runs. The main purpose of these test runs was to obtain operating time on the engine mount. At intervals the engine was shut down and the mount removed for zyglo inspection to determine if any cracks or defects had developed. These inspections disclosed no cracks or defects resulting from engine operation.

A battery life test was performed in conjunction with these first test runs during run no. 3. Beginning with fully charged batteries, the engine was operated continuously at an assumed engine cruise speed of 3600 rpm until the engine stopped due to battery discharge. This particular run lasted for three hours and fifty minutes. The remaining static tests were conducted with a standard automotive type battery with a higher ampere-hour rating.

During runs 1 ough 8 the following conditions were the cause of unsatisfactory engine operation.

1) Spark plug fouling occured during run no. 5 and caused complete stoppage of engine operation. A qualitative chemical analysis made of the residue that fouled the plugs indicated that the type of oil used in the fuel mixture was partially responsible. The use of Shell outboard motor oil was changed to Shell 30W non-detergent oil. No subsequent plug fouling occurred.

2) Ignition breaker arm follower wear occured during run no. 6. Since this condition prevented the breaker points from opening, the spark plugs were unable to ignite the fuel-air charge. The cam was given additional polish to remove any surface imperfections and a thin film of high temperature, silicone base, lubricant was applied to the cam surface to correct this condition.

3) Arcing to ground of the secondary electrical circuit occured at the ignition coil connections. Since this condition prevents one or more spark plugs from firing, engine starting is difficult and if occuring during engine operation causes roughness and loss of power. It was determined that this condition was caused by improper mounting of the ignition coils. The clearance provided between the secondary terminals of the coils and adjacent metal parts is insufficient.

After the completion of run no. 9 the Model H-59A engine was received and installed. The engine mount test runs were continued and at the end of run no. 11 the air pump was coupled to the engine crankshaft to observe the ability of the engine to drive the air pump and to check the adequacy of the pump lubrication

system. In mounting the pump on the engine and driving it directly from the engine crankshaft no unsatisfactory torsional vibration effects were observed. A vibration dampening effect is provided in the pump drive by the use of a flexible coupling between the main shaft of the pump and its splined drive. The pump oil supply system provided adequate lubrication for the pump.

4. Conclusions:

a. Static tests conducted by the engine manufacturer indicated that under static conditions the maximum brake horsepower obtained with the 47-inch diameter propeller was 38.25 at an engine speed of 3625 rpm. Engine cooling was satisfactory using this propeller.

b. Engine operation had no adverse effects on the engine mount structure.

c. The air pump installation operated satisfactorily. No difficulty was experienced with the pump drive assembly and adequate lubrication of the pump was provided by the oil supply system.

d. The engine should be able to operate in excess of four hours on one pair of batteries under standard atmospheric conditions. Low ambient temperatures will reduce battery life as will operation at low power.

e. The selection of oil to be used in the fuel-oil mixture is an important factor in proper engine operation. The ambient air temperatures anticipated during engine operation should also be considered in this selection. It is believed that the very low cooling air temperature encountered was responsible for serious

plug fouling during the static test runs. Low operating temperatures cause excessive carbon build-up in the combustion chambers of the cylinders and also form deposits on the spark plugs.

f. The air supply system operates satisfactorily on the ground. Since no automatic pressure regulation is incorporated in the system to compensate for altitude pressure changes, low internal pressures will be encountered when descending from altitude.

g. The induction system is vulnerable to impact icing which would require that operation under atmospheric conditions facorable to this type of icing be avoided. The system is less vulnerable to fuel-evaporation and throttling icing due to low air consumption of the engine, small mixture temperature drop, inability of water vapor to condense, construction of the air filter and the addition of oil to the fuel.

5. Power Plant Recommendations:

a. It is recommended that the following items be considered in regard to power-plant installation.

1) That more suitable control systems be developed for the installation. The single wire-flexible housing type presently used for throttle and air valve control was adequate but is subject to high friction loads and failure when bends of small radii are used. It is recommended that push-pull rod controls be used on the mount structure and a two-wire flexible cable system be used between the mount and the cockpit.

2) That the air supply system be analyzed to determine pressure regulation requirements, pressure losses and pump requirements in order to provide a more effective system.

b. In regard to powerplant operation it is recommended that environmental testing be conducted to determine oil requirements for low temperature operation, engine cooling at high cooling air temperatures, and carburetor icing possibilities. It is also recommended that flight tests be made to determine the maximum altitude that can be reached with the H-59A powerplant.

TABLE 2 H-59A ENGINE PERFORMANCE

A. ENGINE BREAK-IN RUN (42" Baromotive Propeller)

Time	RPM	Cylinder Head Temp. (°F) 1	2	3	4	OAT	H.P.
2:00	3150	345	200	325	325	56	
1:15	3200	340	295	320	320	62	
:30	3950	425	390	370	370	63	
:15	3950	415	370	370	360	63	42.5

B. ENGINE HORSEPOWER & FUEL CONSUMPTION (47" Goodyear Aircraft Propeller)

Time	RPM	Cylinder Head Temp. 1	2	3	4	OAT	H.P.
:45	3625*	370	375	400	440	60	38.25
:30	3500	360	365	390	435	60	33.75
:30	3000	295	295	325	350	61	19.50
:30	2000	225	215	270	280	61	5.00

*Maximum rpm reached

Notes:

1. The runs reported under item A were made as a pusher installation. Runs reported under item B were made as a tractor installation.
2. Maximum horsepower obtained under item A was 42.5.
3. Due to propeller design full engine power was not obtained under item B.

TABLE 3
H-59 & H-59A-ENGINE PERFORMANCE

Run	Date	RPM	CHT(^{o}F)		OAT(^{o}F)	Time	Remarks
			3	4			
1	12-29-56	1200-4000	INOP	212	20	:20	H-59 engine, 42" prop., pump installed but not coupled
2	1-2-57	3200	INOP	176	20	:20	
3	1-3	3600	INOP	176	15	3:50	Battery life
4		3600	INOP	162	25	:30	
5	1-4	3600	INOP	122	20	:45	Spark plugs fouled
6		3600	INOP	158	20	:30	Breaker arm follower worn
7		3600	INOP	160	30	:30	Coil mounting failed. Engine mount removed for zyglo inspection
8	1-14	3500	INOP	214	15	1:30	Engine oil type changed to 30W, non-detergent
9	1-15	3500	INOP	212	16	6:30	Engine mount removed for zyglo inspection
10	2-6	3000	240	302	31	1:00	H-59A engine installed, 42" prop. pump installed but not coupled
11		3200	258	312	30	1:00	
12		3500	284	328	30	1:00	Pump coupled to engine drive
13		3800	302	347	30	:30	
14		4000	258	328	30	:15	
15		3600	275	320	30	:15	Engine mount removed for zyglo inspection
16	2-12	2500-4000	258	284	25	:15	Engine installed on airplane for pressure test
17	2-19	3000	238	284	25	:15	Now pump installed for break-in
18		3500	275	338	25	:15	
19		3500	275	338	25	:05	
20		4150	302	292	25	:05	Engine mount removed for zyglo inspection

FINAL WEIGHT AND BALANCE

Table 4 represents the actual weight and balance breakdown of the airplane as it was flown in the flight test program.

Table 4 (Sheet 1)
FINAL ACTUAL WEIGHT AND BALANCE

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Wing			42.4	(86.6)		3672	(55.8)		2365
Envelope		26.0		80.5	2093		56.2	1461	
Ailerons (2)		3.8		108.0	413		50.7	193	
Flap		3.2		108.8	348		50.7	162	
Brace Wire		2.0		94.4	189		68.8	138	
Patches (8)		2.5		81.0	203		56.2	141	
Hinges		1.2		104.9	126		52.0	62	
Air (Δ) at 7 psi		3.7		81.0	300		56.2	208	
Tail			11.8	(233.1)		2750	(60.3)		712
Vertical		4.9		231.0	1132		71.4	350	
Fin	3.1								
Rudder	1.6								
Hinge	0.2								
Horizontal		5.7		235.3	1341		51.7	295	
Stabilizer	3.0								
Elevator	2.4								
Hinge	0.3								
Brace Wire		0.3		230.0	69		53.0	16	
Patches		0.5		230.0	115		53.0	27	
Air (Δ) at 7 psi		0.4		233.0	93		61.0	24	
Body			34.0	(73.6)		2502	(36.0)		1224
Basic Structure		11.4		136.5	1556		39.1	446	
Envelope	10.0								
Air (Δ) at 7 psi	1.4								
Secondary Structure		22.6		(41.9)	946		(34.4)	778	

Table 4. Final Actual Weight and Balance (Sheet 2)

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Canopy	5.4			30.0	(162)		49.0	(265)	
Windshield	0.7			30.0	(21)		56.0	(39)	
Seat	13.6			46.6	(634)		28.6	(389)	
Fairing	0.4			46.6	(19)		28.6	(11)	
Assembly Straps	1.0			46.6	(47)		28.6	(29)	
Air (Δ) at 7 psi	0.5			43.0	(22)		32.0	(16)	
Safety Belt	1.0			41.3	(41)		28.6	(29)	
Alighting Gear - Unicycle			14.1	(73.0)		1030	(22.6)		318
Main		10.5		65.0	682		20.1	211	
Wheel, Tire, Tube, Axle & Air	5.3			64.0	(339)		16.3	(86)	
Shock Structure	5.2			63.0	(343)		24.0	(125)	
Tail Skid		0.6		231.3	139		34.0	20	
Wing Tips (2)		2.5		81.0	203		30.0	75	
Nose Skids (2)		0.5		12.0	6		23.0	12	
Surface Controls			6.6	(54.2)		358	(43.3)		288
Stick and Support		0.9		20.9	19		32.0	29	
Pedals and Support		0.8		11.0	9		31.0	25	
Cables and Fittings		4.4		68.0	299		46.0	202	
Scuff Patches		0.5		61.4	31		60.0	30	
Engine Section			10.5	(108.2)		1136	(65.9)		692
Engine Mount		6.9		108.1	746		65.1	449	
Saddle Assembly		2.1		100.0	210		56.0	118	
Accessory Shroud		1.5		120.0	180		83.0	125	
Propulsion Group			69.5	(106.6)		7407	(80.9)		5620
Engine Installation		57.2		109.7	6275		83.8	4796	

Table 4. Final Actual Weight and Balance (Sheet 3)

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Engine	51.1			110.0	(5621)		83.0	(4241)	
Carburetor Fuel Pressure Regulator Hose	6.1			107.2	(654)		91.0	(555)	
Engine Controls		0.4		108.0	43		83.0	33	
Fuel System		4.1		75.9	311		35.1	144	
Tank	3.1			74.5	(231)		30.2	(94)	
Plumbing	1.0			80.0	(80)		50.0	(50)	
Propeller and Hub		7.8		99.7	778		83.0	(647)	
Pneumatic Group			13.5	(121.5)		1640	(82.9)		1119
Compressor		10.2		120.8	1232		83.0	847	
Relief Valve		1.0		129.4	129		83.0	83	
Selector Valve		0.6		127.2	76		86.0	52	
Oil Separator		1.1		119.7	132		77.0	85	
Oil Tank		0.6		117.6	71		87.0	52	
Electrical Group			13.0	(118.2)		1537	(79.4)		1032
Battery (2)		8.0		126.5	1012		83.0	664	
Battery Case		1.3		126.5	164		83.0	108	
Main Electric Cable		1.7		75.0	128		59.0	100	
Coil Assembly		2.0		116.3	233		80.0	160	
Instruments			8.5	(34.1)		290	(36.6)		311
Pressure Gage		0.4		20.0	8		46.0	18	
Airspeed Indicator		0.6		20.0	12		46.0	28	
Pitot Tube		0.1		4.1	0		33.0	3	
Altimeter		1.4		20.0	28		46.0	64	
Engine Test Instruments		6.0		40.4	242		33.0	198	

Table 4. Final Actual Weight and Balance (Sheet 4)

ITEM 1 2 3	WEIGHT 3	WEIGHT 2	WEIGHT 1	HORIZ DIST	HORIZONTAL MOMENT 2	HORIZONTAL MOMENT 1	VERT DIST	VERTICAL MOMENT 2	VERTICAL MOMENT 1
Weight Empty			223.9	(99.7)		22322	(61.1)		13679
Pilot			200.0	41.3		8260	37.2		7440
Gross - No Fuel 24.0% MAC			423.9	(72.1)		30582	(49.8)		21119
Fuel 15 Gallon at 6.2#/Gallon			93.0	74.5		6929	30.2		2809
Gross - With Fuel 24.8% MAC			516.9	(72.6)		37511	(46.3)		23928

L.E. Wing at F.S. 57.7

Section 4

PHASE IV

FLIGHT TESTING

GER 8146

Bruce K. Bain
John W. Phillips
J.T. Blair

Original estimates indicated that the Inflatoplane could not be completed until March 1, 1957, and funds would not be sufficient to make more than one or two flights to show the airworthiness of the plane. However, final assembly was about three weeks ahead of schedule and some money was available so that the flight testing program was expanded so that limited performance data could be obtained and some improvements could be made. A total of 64 flights were made varying from 1 to 32 minutes in duration. Six and one-half hours of flight time were logged with a total of 50 hours of airframe time in testing, taxiing, and actual flying at the 7 psi inflation pressure. Flight testing was conducted at Wingfoot Lake Airship Base between February 12 and March 16, 1957. One flight was made from Akron Municipal Airport to Wingfoot Lake Airship Base.

INITIAL FLIGHTS

Initial flights were conducted on February 12 and 13, 1957, with the tricycle landing gear after several high speed taxi runs. A 42-inch propeller was used at first to give greater propeller clearance. These flights are summarized below in Table 5.

No. of flights	8
Max. altitude	20
Distance	200 - 600 feet
Gross weight	467 lbs
C. G.	25.8% mac
Take-off speed	32 mph
Max. speed	51 mph
Flight time	3 min
Taxi time	2 hrs

TABLE 5. Initial Flight Tests

During these tests it was noted that there was no aileron control below 32 mph, which made smooth take-offs and landings difficult. Close inspection of the control stick and aileron movements showed that only about half the necessary movement of 15 degree down and 25 degrees up was being realized. It was found that the control forces were causing deflections in the structure and at the teflon cable guides because of friction in the guides and routing of the cables. Also, the angle of incidence of the wing was too high and it was possible that the ailerons were stalled out much of the time. The high angle of incidence of the wing resulted in poor take-off performance and most take-offs were actually made off the nose wheel. Also, it was necessary to keep the stick well forward to maintain level flight. Since the nose wheel was almost directly under the pilot and was in contact with the ground during the entire ground run the pilot noticed quite a pounding on the cockpit in taxiing. The angle of incidence was measured after these flights and found to be 13 degrees instead of the 10 degrees as designed because the engine weight depressed the trailing edge of the wing. There appeared to be a slight drooping at the trailing edge of the tips due to the long unsupported length. This undoubtedly contributed to stalling of the ailerons. With the empennage brace wires at the tips of the leading edges large deflections were noticed in the vertical stabilizer from the ground, although the pilot felt that rudder control was adequate at all times. At the conclusion of these initial flights the following modifications were accomplished:

SECOND FLIGHT TEST SERIES

1. The teflon cable guides in the aileron control system were replaced with

pulleys and the cables were rerouted. These modifications restored proper aileron control movement.

2. The angle of incidence of the wing was reduced to 8-1/2° by cutting down the bulkhead pad and lowering the leading edge. An extra pair of brace cables was added on the upper side of the wing near the trailing edge to control the drooping tendency.
3. The rear wheels of the tricycle landing gear were lowered 1.5 inches to give a better take-off attitude. This change was made simply by changing the relative lengths of the positioning straps.
4. The empennage bracing system was revised so that one set of cables connected the mid-span of the landing edges and another set connected the hinge lines at the 2/3 span points.

After completing these modifications a second series of tests was conducted on February 20, 21, and 22, 1957, to altitudes of 300 feet and were kept within the flight pattern of the field. The change in the wing incidence and the landing gear angle gave the proper attitude to the airplane in take-off and landing (fig. 27). Aileron control was considerably improved, although additional control was still desirable. A forward stick position was required above 38 mph, indicating that the wing incidence could be reduced further. The empennage was free of all visible deflections with the new bracing arrangement. Tests made to determine take-off and landing distance, maximum, cruise, take-off, and approach speeds, and maximum engine RPM are summarized in Table 6 below.

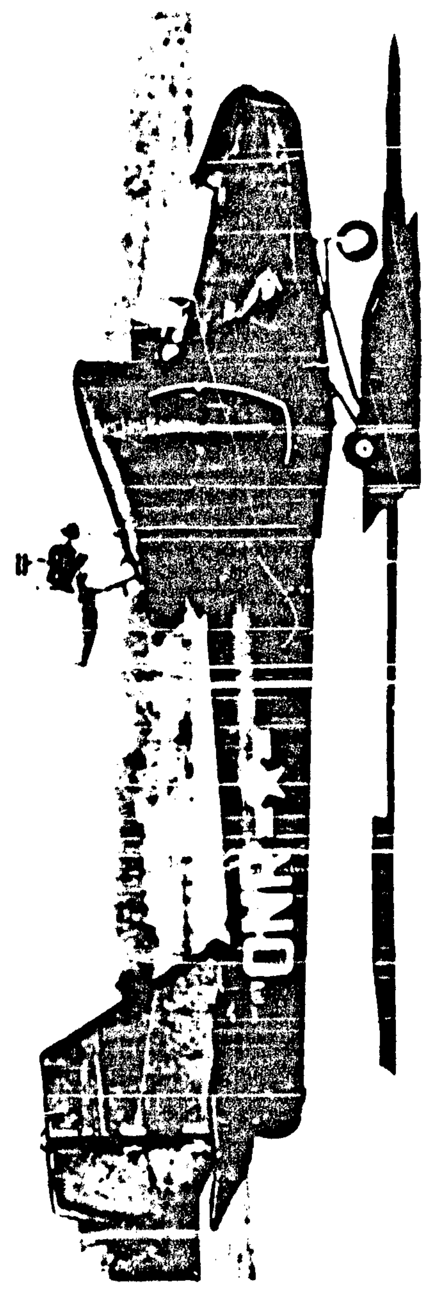


Figure 27. Inflatoplane Landing on Tricycle Gear

Propeller - 42" dia. x 12.5° pitch
Tricycle gear

Gross Wt. Lbs.	TO Dist Ft.	TO V MPH	V max IAS MPH	Cruise V MPH	V max RPM	Approach V min. MPH	Land Dist Sod FT.
470	190 HS*	31	59	50	-	34	231
490	203 HS	32	58	50	4400	35	-
490	229 HS	32	-	-	-	-	-
540	283HS	34	56	50	-	40	-
540	288 HS	34	-	50	-	40	387
540	350 S*	34	-	50	-	-	275

*HS Hard Surface
* S Sod

Table 6 Flight Tests with Tricycle Landing Gear

The tricycle landing gear was then removed and the single wheel installed. A series of tests similar to those above was conducted (fig. 28). Table 7 summarizes the unicycle tests.

Propeller - 42" dia. x 12.5° pitch
Unicycle gear

Gross Wt. Lbs.	TO Dist ft.	TO V mph	V max IAS mph	Cruise V mph		Approach V min. mph	Land Dist ft.
435	180 HS	31 mph	64 mph	50	3700 RPM	34 mph	380
435	200 HS	31	-	52		-	230
485	300 S	32	-	52		-	-
485	200 S	32	-	52		34	275

Table 7 - Flight Tests with Unicycle Landing Gear

Figure 29 shows the take-off and landing distances of the airplane for both landing gear vs. gross weight, corrected to standard conditions. It may be seen that the take-off distances for both landing gears are practically the same for the same gross weight. The gross weight with the unicycle gear is, of course, less than

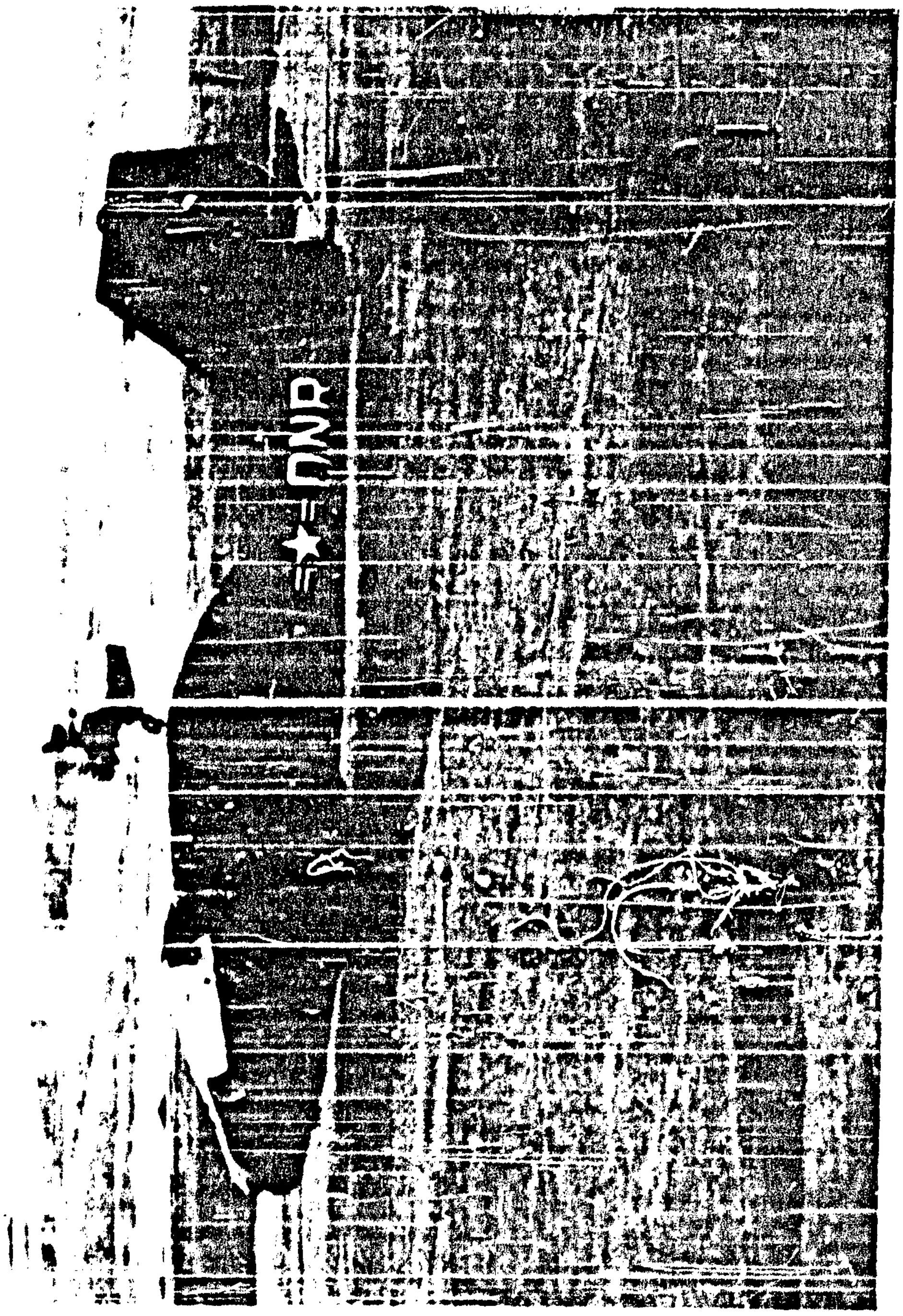


Figure 28. Inflatoplane Larding on Unicycle Gear

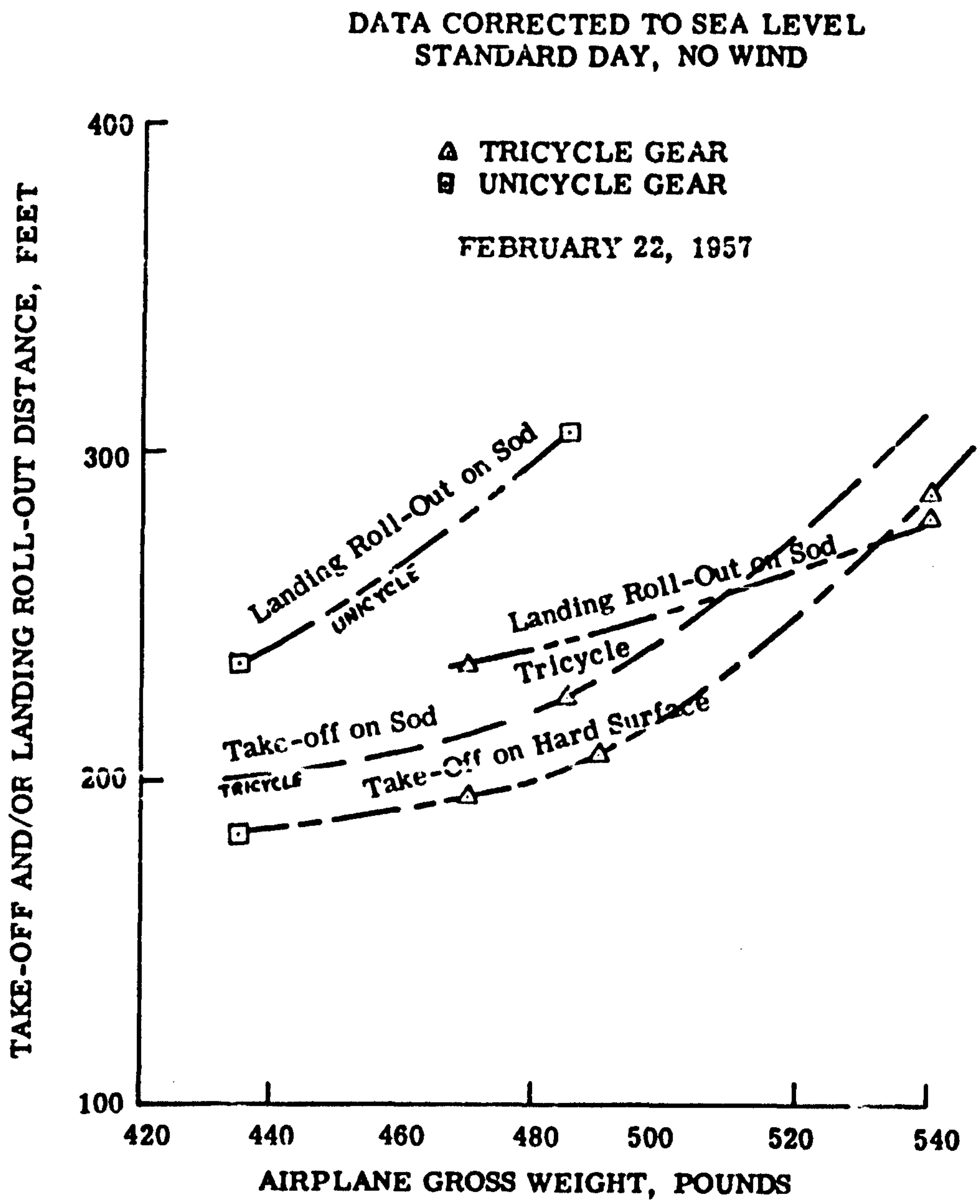


Figure 29. Take-Off Distance vs Gross Weight

with the tricycle gear. With the lower drag the top speed with the unicycle gear is 5 mph faster than with the tricycle gear. A total of 30 flights were made during these tests. Several minor changes were made at this time. A relief valve specifically designed for use with this compressor was included in the air line to guard against over-inflation. After the bulkhead pad was cut down there was an opening directly behind the pilot's head which causes a low pressure to develop inside the cockpit. This caused the canopy windshield to collapse down against the pilot's helmet. The opening was closed with a curtain and the windshield assumed its proper inflated position in later flights.

CLIMB TESTS

On March 4, 1957, climb tests were conducted with 42-inch diameter x 12.5 degree pitch and a 47-inch diameter x 12.5 degree pitch propellers to determine the rate of climb and service ceiling of the airplane. This data is summarized in Table 8 below.

Table 8. Climb Tests with 42 and 47 Inch Propellers

Propeller	Altitude at lapsed time, minutes						Gross Wt.	Climb Airspeed
	0	.5	1.0	1.5	2.0	2.5	Lbs	IAS mph
47"	1200	1620	1950	2340	2640	3000	459	42
47	1200	1630	1960	2300	2650	3000	456	42
42	1200	1550	1900	2240	2520	2800	453	42

Full throttle climb 3500 RPM with 47 inch propeller.
Engine temperature 160°C.

Figure 30 shows this data plotted to give the rate of climb at different altitudes, service ceiling and absolute ceiling at a gross weight of 459 lbs. These tests show that the 47-inch propeller is better than the 42-inch propeller, as anticipated.

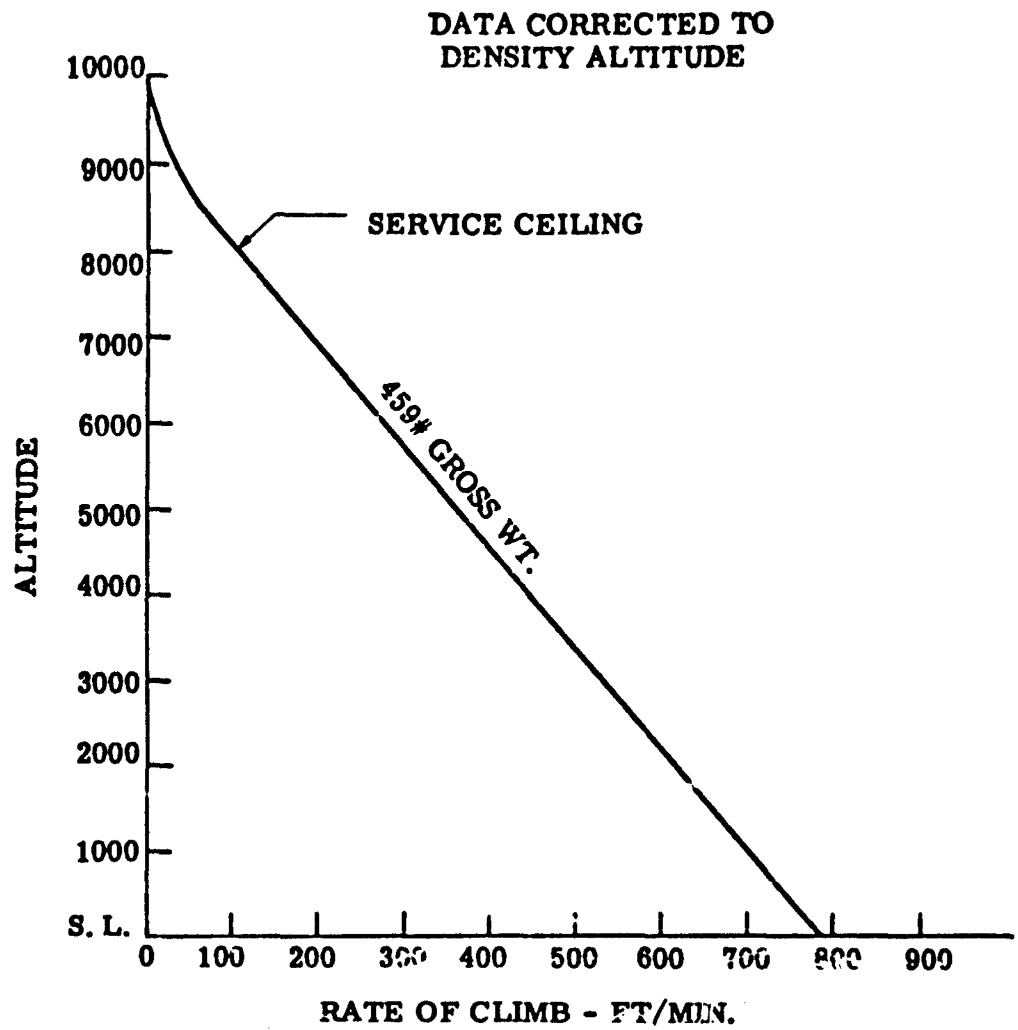


Figure 30. Rate of Climb vs Altitude

With the exception of a fuel comsumption test no specific powerplant flight tests were made. However, during the airplane flight tests observation of engine operation was made in regard to engine cooling and air system operation.

ENGINE OBSERVATIONS

1. Fuel Consumption: A flight test was made to determine fuel consumption. This test was conducted by fueling the airplane with a known weight of fuel and then by weighing the remaining fuel drained from the tank after the test flight. Weight of the fuel at the start of the test was 29 pounds and at the end of the test was 19 pounds indicating that 10 pounds of fuel were used. The test flight lasted approximately one-half hour at an engine speed of 3150 rpm and a flight speed of 57 mph. Fuel consumption was then determined by dividing the weight of fuel used per hour (2 x 10 pounds) by the weight of one gallon of fuel (6.35 pounds). This results in a fuel consumption of 3.15 gallons per hour which is in approximate agreement with the fuel consumption curve established by the engine manufacturer in figure 25 and gives a range of 470 miles.

2. Engine Cooling: No excessive cylinder head temperatures were observed at any time during the airplane flight tests. It was observed, however that during climb at full power cylinder head temperatures reached 400°F. Since cooling air temperature was approximately 35°F., the maximum allowable cylinder head temperature of 450°F will probably be exceeded if climb is made at full power when outside air temperature exceeds 85°F. Cylinder head temperatures at engine cruise speeds of 3000 - 3200 rpm reached a maximum of 330°F. which

would allow ample temperature rise before the maximum cylinder head temperature limit is reached.

3. Air System Operation: During the airplane flight tests it was observed that the maximum internal airplane pressure that is obtained before take-off could not be maintained at altitude. This condition was alleviated by increasing the reflief setting of the valve. The present system does not automatically compensate for differences in atmospheric pressure at altitude. Therefore it would be necessary for the pilot to resupply air when descending to lower altitudes. However, since the maximum operating altitude may be restricted to less than 4000 feet due to carburetor limitations the resultant pressure loss should not seriously affect airplane performance.

ADDITIONAL FLIGHT TESTS

Six additional flights were made on March 12, 13, and 16, 1957, for demonstration purposes. The maximum altitude flown to date was 3500 feet MSL. Flights have been made in winds up to 48 mph and the plane has been landed and taken off in gusts as high as 20-30 mph. Inflation pressure has been as low as 5 psi and as high as 7-3/4 psi. The maximum inflation pressure of the plane was 9-1/2 psi. During the endurance flight an accelerometer was installed in the airplane to measure maximum and minimum G loadings. The measured maximum load was 2.6 G's and the minimum 0 G's.

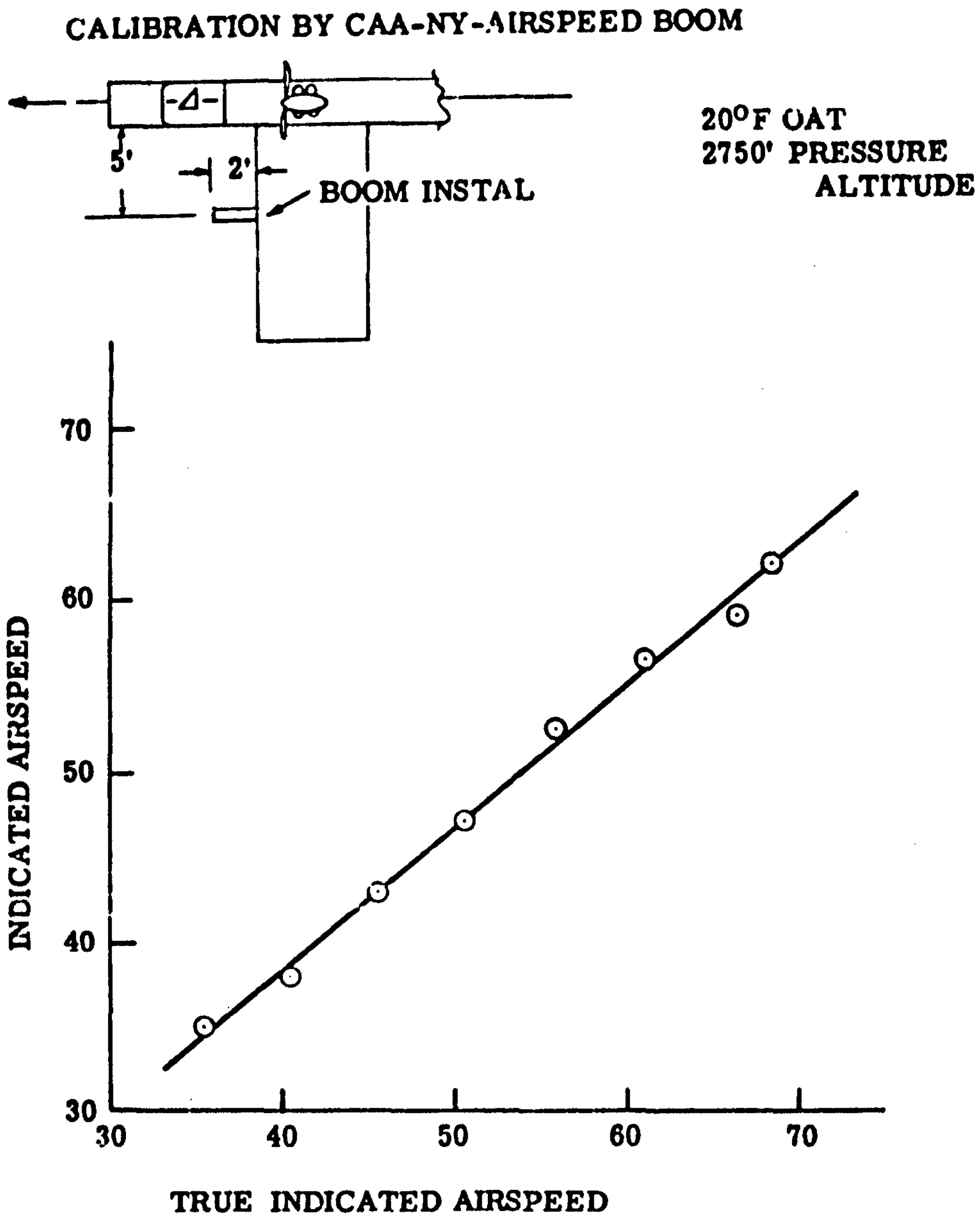


Figure 31. Airspeed Indicator Calibration

AIRSPEED CALIBRATION TESTS

On March 5, 1957, a series of flights were made with a CAA-NY calibrated airspeed boom to calibrate the airspeed indicator on the airplane. Maximum speed runs were also made at two different altitudes. This data is summarized in Table 9 below.

Slip Indicated Airspeed-mph	35	38	43	47	52.5	56.5	59	62
CAA Boom Indicated Airspeed, Run #1-mph	35	40	45	50	55	60	65	67
CAA Boom Indicated Airspeed, Run #2-mph	35.5	40.5	45.5	50.5	55.7	61	66.2	68.3

At a true/indicated airspeed of 68.3 mph in level flight at a pressure altitude of 2750 feet and an outside air temperature of 20°F., the true airspeed was 69.1 mph or 60.1 knots.

In a speed run in level flight at a pressure altitude of 1100 feet the indicated airspeed was 65 mph, which in a true airspeed of 71.5 mph or 62 knots.

Table 9 Air Speed Indicator Calibration

Figure 31 shows the calibration of the airspeed indicator. The top speed of 71.5 mph or 62 knots could be improved by fairing in the area behind the bulkhead and improving the trim of the airplane by changing the wing angle of incidence.

Section 5

DISCUSSION OF RESULTS

GER 8146

Bruce K. Blair
J.T. Blair

In general, the results obtained with the Inflatoplane were excellent. Performance was considerably better than anticipated, particularly with respect to take-off distance and range, which are two of the most important performance features in this application. Depending on the gross weight, the take-off distance was as short as 180 feet. Endurance was over 8 hours and maximum range about 470 miles, which was double the initial requirement. Indications are that this range could be increased by 50% by merely increasing the capacity of the fuel cell. The calculated service ceiling was 8000 feet and the absolute ceiling around 10,000 feet. However, present engine carburation would probably reduce this value. The engine was reliable and the airplane was flown in adverse weather, showing the plane's over-all usefulness.

STRUCTURAL ASPECTS

Structurally the airplane was basically sound. Several modifications were indicated, however, which would improve the over-all structure. To increase the full cross-sectional area of the wing allowance should be made for shrinkage in the curing process.

As dacron materials become available they should be considered for use in the fuselage and other members as a means of increasing stiffness. There were a number of uncertainties concerning actual stress conditions once initial deflections had taken place. These uncertainties could be largely eliminated by performing full scale wind tunnel tests. Of particular interest would be tests showing the behavior of the airplane at or near buckling conditions.

FLIGHT TESTS

Flight tests indicated the desirability of a few changes which would simplify or improve over-all operation. The canopy was modified several times to find the best arrangement for pilot entrance and exit and should be investigated further. The landing gear used was designed to give minimum aerodynamic drag, weight, and packaged size and it functioned properly on the hard surface and sod runways. There were indications, though, that the landing gear should be lowered and the wheel diameter increased for rough field operation by inexperienced pilots. The air pressure control system was manually controlled and after experimentation operated satisfactorily. An automatic control system might be desirable on future Inflatoplanes.

PACKAGING AND GROUND HANDLING

Although packaging and ground handling have not been fully investigated, preliminary studies indicate that the airplane can be moved reasonable distances and erected by one man. Figure 32 shows a preliminary packaging and handling arrangement. The pallet was 38 inches wide x 54 inches long and height from the pallet to the uppermost projection (the accessory shroud) was 40 inches. As shown in the illustration, the package was handled as a wheelbarrow and there was adequate space left for an inflation bottle, gasoline, an emergency hand pump, and any other necessary accessories.

The package dimensions were controlled completely by the engine d mount as all inflated components were packed around these two items. The height

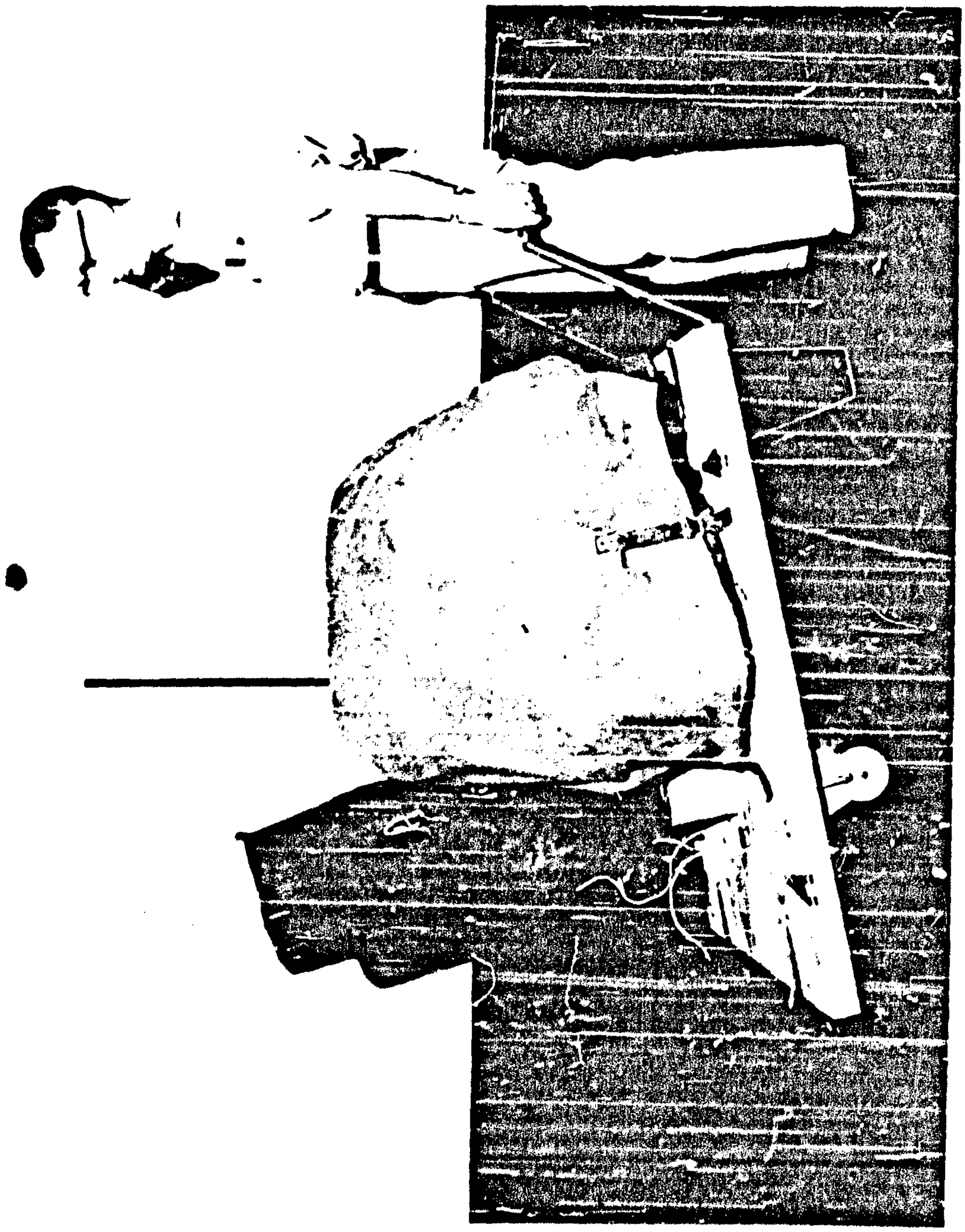


Figure 32. Airplane Packaged for Transport

dimension could easily be reduced from 40 to 32 inches by relocation of the batteries, which are in the back of the accessory shroud. It was felt that the packaged size could be reduced still further by relocation of many of the engine accessories. The airplane has been packaged with the gasoline inside the fuel cell, eliminating the need for separate gasoline containers and simplifying preparation for flight. A hand pump for inflating large fabric liferafts was procured to determine the feasibility of inflating by hand if a compressed gas cylinder was not available. This pump had a 6 inch diameter and 15 inch stroke.

It is believed that one man could inflate the airplane to 3 psi in 10 minutes. The engine compressor would be used to complete inflation to 7 psi in an additional minute. The engine has been run with the inflation pressure as low as 2 psi. Inflation with bottled gas has been estimated to take approximately one minute. The total amount of time to prepare the airplane for flight should not exceed 3 minutes, with the fuel in the cell. A complete report covering packaging and ground handling will be prepared as part of the Two-Place Inflatoplane Development Program.

WEIGHTS

The final weights breakdown showed that the net weight of the airplane as it was flown in the flight test program was 223 pounds with the unicycle gear and 245 pounds with the tricycle gear. This weight included 8.6 pounds for flight test instruments and 6.5 pounds for inflation air. At the end of the program it was learned from the pump manufacturer that lubrication was not required on their pump in runs

up to 8 hours. The weight of the oil supply, separator, and hosing was 1.7 pounds. Bullet hole tests on the wing panel (Appendix A) showed that a smaller compressor could be used, with a added weight saving of 5.5 pounds. The H-59A (44HP) engine used in this program weighed 9.2 pounds more than the 40 HP H59 engine proposed.

POWER PLANT

The H-59A engine is an improved version of the H-59 engine. Our experience has shown that both engines are reliable and that the lighter engine could be used successfully if it is felt that the weight saving is significant. Including all the weight savings mentioned, the net flying weight of the airplane could be reduced to 198 pounds without affecting performance or safety to any noticeable degree. In the packaged condition without the inflation air the weight would be 191.5 pounds. If absolutely necessary the dry weight could be reduced still further by removing the air compressor system (9.6 pounds), the canopy (5.8 pounds), one battery with its mounting brackets (5.3 pounds), and the two seat adjustment pads (9 pounds). Removal of these items would reduce the deflated weight to 169.9 pounds. However, the removal of the air compressor is not recommended. Several areas for weight reduction is the pneumatic components have been noted which could be incorporated in future Inflatoplanes.

Section 6

CONCLUSIONS AND RECOMMENDATIONS

GER 8146

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CONCLUSIONS

It may be concluded that the Inflatoplane developed in this program can readily accomplish the mission of escape and evasion outlined in the Introduction. Specifically, the airplane met or exceeded the contract design requirements. Take-off performance and range were outstanding, making the airplane exceptionally good as a long or short range escape device. Preliminary packaging studies indicate that the airplane can be dropped by parachute, easily moved, and quickly prepared for flight by one man.

RECOMMENDATIONS

The following recommendations are made for future work on the Inflatoplane:

1. Full-scale wind tunnel tests on the present airplane should be made to determine actual wing loadings, deflections of the structure under various loads, and flutter and buckling characteristics.
2. Additional engineering work should be done to make the improvements indicated in the Discussion of Results.
3. Additional, modified Inflatoplanes should be made for field evaluation.
4. Additional capabilities and uses of the Inflatoplane should be investigated, including, seaplane and amphibious operation.
5. Other applications of pneumatic structures in the aircraft field should be considered.

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Section 7

APPENDIX A

TESTS TO DETERMINE AIR LOSS

THROUGH BULLET HOLES IN "AIRMAT"

GER 8146

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BULLET HOLE TESTS ON "AIRMAT"

A flat, 3 inch Airmat panel was inflated to 6-1/2 psi (the operating pressure of the airplane) and a .38 caliber bullet was fired through it. A Flowrator was placed in the supply line to the Airmat and the flow rate necessary to maintain different internal pressures was measured. A Bourdon type pressure gage was used to measure the internal pressure of the Airmat and a Foster regulator was used to regulate the flow rate through a clean hole (figure 33) to determine the coefficient of discharge, C_D, at different pressures (figure 34). The increase in C_D is due to the increase in size of the hole under pressure. Using these values for C_D figure 35 was drawn showing loss of air through a .30 caliber bullet hole for different internal pressures.

During the flight test program a test was conducted to determine the actual pressure loss in the wing from five .30 caliber bullet holes, and the ability of the pump in maintaining pressure. A 4 foot panel of the wing was fabricated, inflated to 7 psi, and punctured by five .30 caliber rifle shots. The panel was connected to the compressor system and the internal pressure in the panel was measured with a pressure gage, at various engine-compressor speeds. The actual volume lost was determined from compressor calibration curves. The test data is summarized below.

No. of holes	Engine Speed rpm	Internal Pressure psig	Actual Volume Loss cfm
5	2000	4.5	23.4
5	2500	5.0	29.3
5	3000	5.5	35.0
5	3600	6.5	42.0

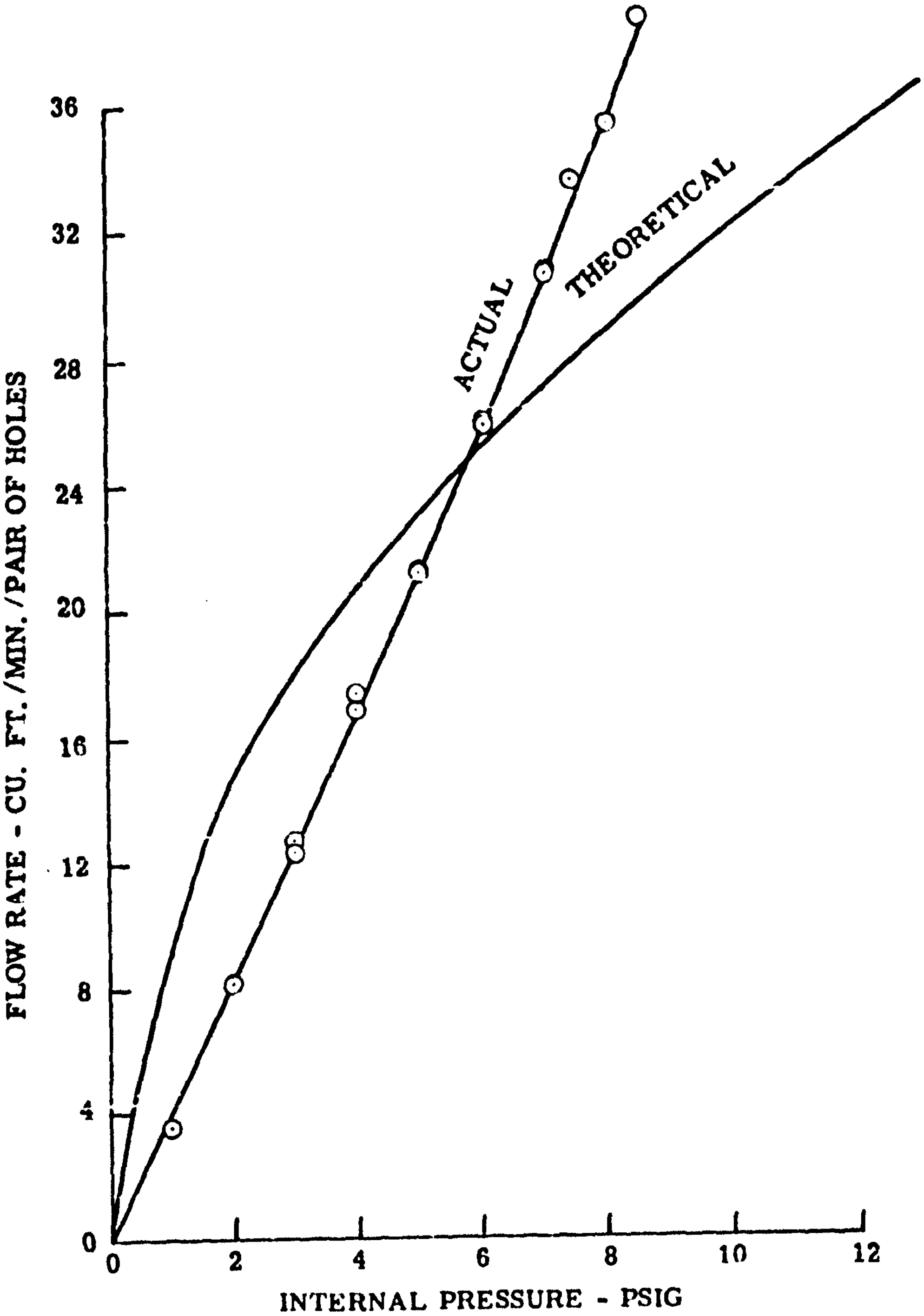

Figure 33. Gas Loss Through Two .38 Cal. Bullet Holes In a Three-Inch Airmat at Different Internal Pressures

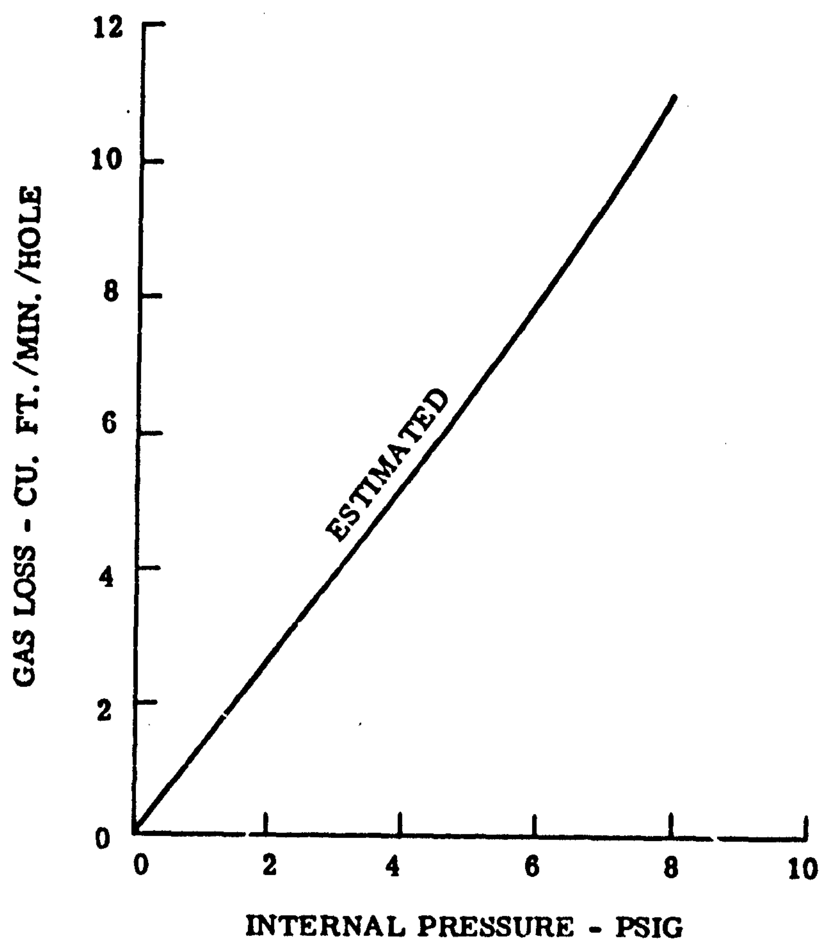

Figure 34 Gas Loss Through A .30 Caliber Bullet Hole in a Three-Inch Airmat at Different Internal Pressures

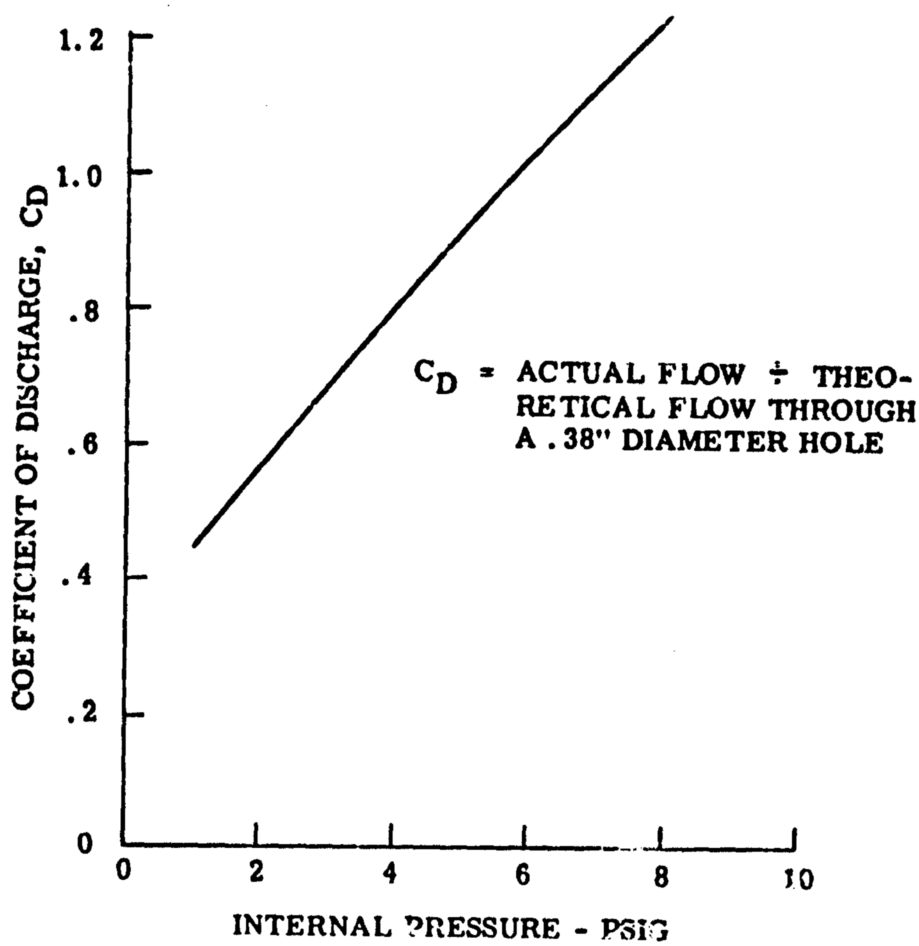

Figure 35. C_D at Different Internal Pressures

This test showed that the air loss from bullet holes was considerably less than anticipated and that a smaller compressor could be used.

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Section 8

APPENDIX B

CARBURETOR ICING

ON THE H-59A NELSON ENGINE

GER 8146

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CARBURETOR ICING

Our experience with the Nelson H-59A engine has indicated no tendency to induction system icing. However, since no provision is made for carburetor heat to prevent or remove ice, consideration should be given to this problem in order to investigate the possibility of carburetor icing. Therefore, this section of the report is presented as a commentary on the subject of carburetor icing as it applies to the Inflatoplane installation and is based on the results of NACA investigations reported in Reference 9 . In general, three types of icing are possible in this installation; impact icing, fuel-evaporation icing, and throttling icing. Each of these types of icing and their probable effect and occurrenceduring engine operation are discussed below.

IMPACT ICING

Impact icing occurs when a sub-freezing surface comes in contact with super-cooled water droplets and can form on external surfaces, duct inlets and walls, and on exposed elements inside the carburetor. Since the Inflatoplane power-plant installation utilizes a very simple induction system consisting of an air scoop and filter mounted directly on and above the carburetor air inlet, impact ice could form within the scoop and on the face of the filter, thereby blocking air flow to the carburetor. Generally, this type of icing occurs when flying through clouds or in a freezing rain. The only method of preventing this type of icing with this installation is to avoid operation under atmospheric conditions that can cause impact ice formation.

FUEL - EVAPORATION ICING

Fuel-evaporation icing occurs when fuel is introduced into the airstream of the carburetor causing a decrease in the air temperature and surrounding metal surfaces. This temperature reduction allows water vapor, if present, to freeze upon contact with adjacent metal surfaces. On the H-59A installation this type of icing could accumulate on the boost venturi and throttle valve of the carburetor and cause engine power failure. It is doubtful that fuel-evaporation icing can easily occur on this installation because of the following reasons:

1. Tests on the original Goodyear Aircraft Inflatoplane indicated a maximum carburetor fuel-air mixture temperature drop of approximately 20°F. This would indicate the possibility of ice forming in the carburetor only at inlet air temperatures below 50°F. However, at these inlet air temperatures, a psychrometric chart shows small amounts of moisture present at high humidities.
2. At the maximum airflow required by the engine of approximately 400 pounds per hour, the actual amount of moisture passing through the carburetor under the above conditions is low.
3. Icing due to water vapor passing into the carburetor is remote due to the amount of heat that is necessary to be given off in order that condensation can occur.
4. The air filter is of such construction that a large part of the water vapor present will be removed from the airstream by the filter.
5. To lubricate the engine, oil is mixed with the fuel. This addition of oil to the fuel lowers the volatility of the fuel thereby reducing the tendency to fuel-evaporation icing.

THROTTLING ICING

Throttling icing is caused by condensation and freezing of water vapor resulting from the expansion cooling of the charge air as it passes through restrictions in the induction system. This type of icing could possibly occur on this installation at the throttle valve when operating at low power settings where the edge of the valve is near the wall of the carburetor. Ice formation at this throttle valve position could block the idle discharge port and cause engine stoppage. The possiblity of this type of ice formation occuring is believed to be remote for the same reasons given for fuel-evaporation icing since both conditions are dependent on the amount of water vapor present in the airstream of the carburetor. In addition, the temperature drop due to expansion cooling is not high since the 20^{o}F. temperature drop observed includes fuel-evaporation and expansion cooling.

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Section 9

APPENDIX C

PHOTOGRAPHIC SUPPLEMENT

ON GA447 INFLATOPLANE

GER 8146

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PREFACE

The photographs in this appendix show the Goodyear Aircraft, Model GA447 One-Place Inflatoplane in flight and demonstrate the packaging and ground handlii sequence. A description of the inflation sequence is found in section V, pages 100 and 101 of this report under the heading "Packaging and Ground Handling."

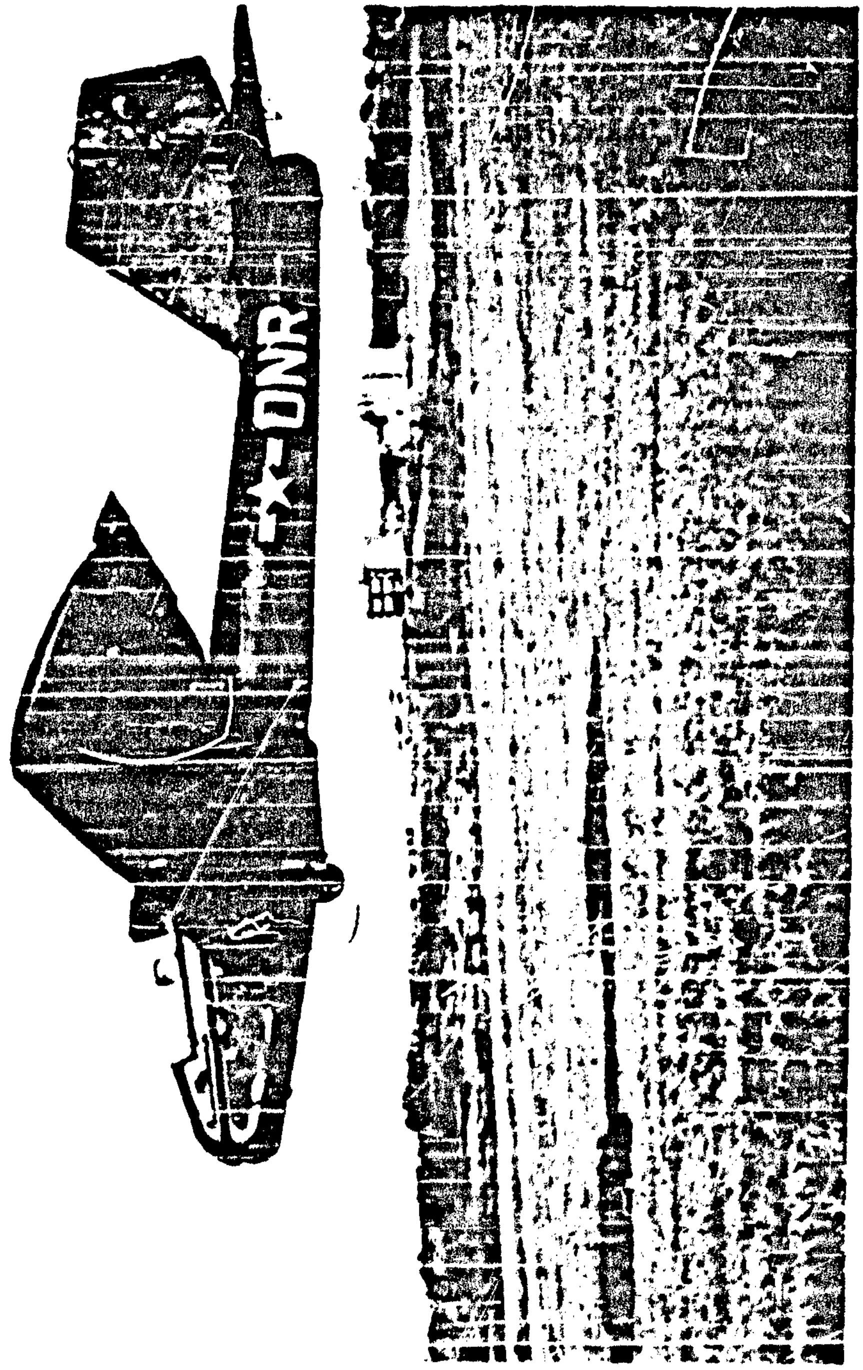

Figure 36. Side View of GA447 Inflatoplane Showing Unicycle Landing Gear

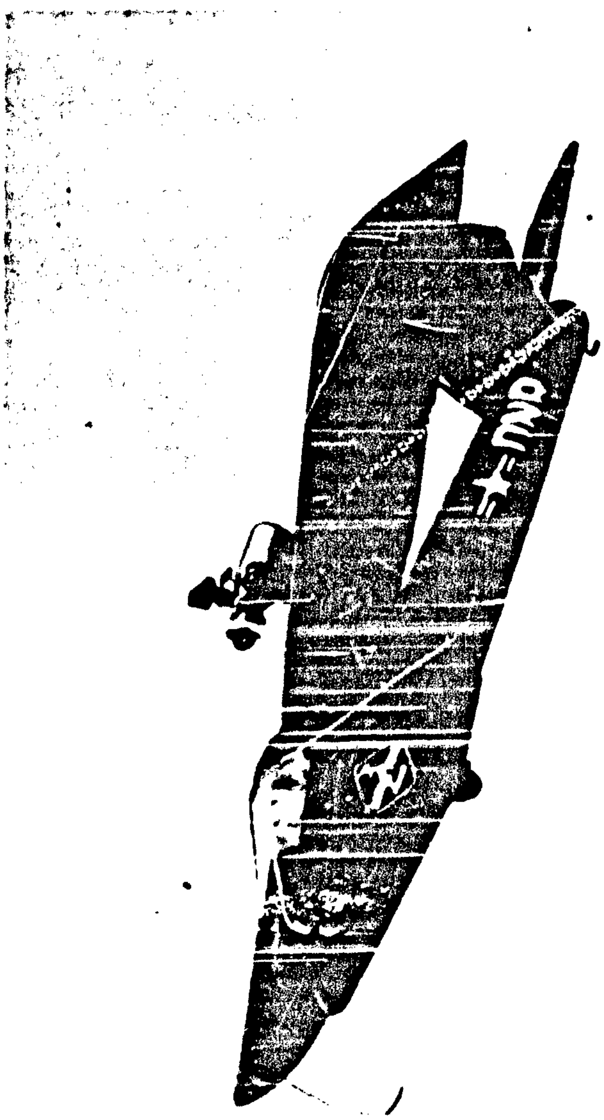

Figure 37. View of GA447 Inflatoplane in Climbing Turn Attitude

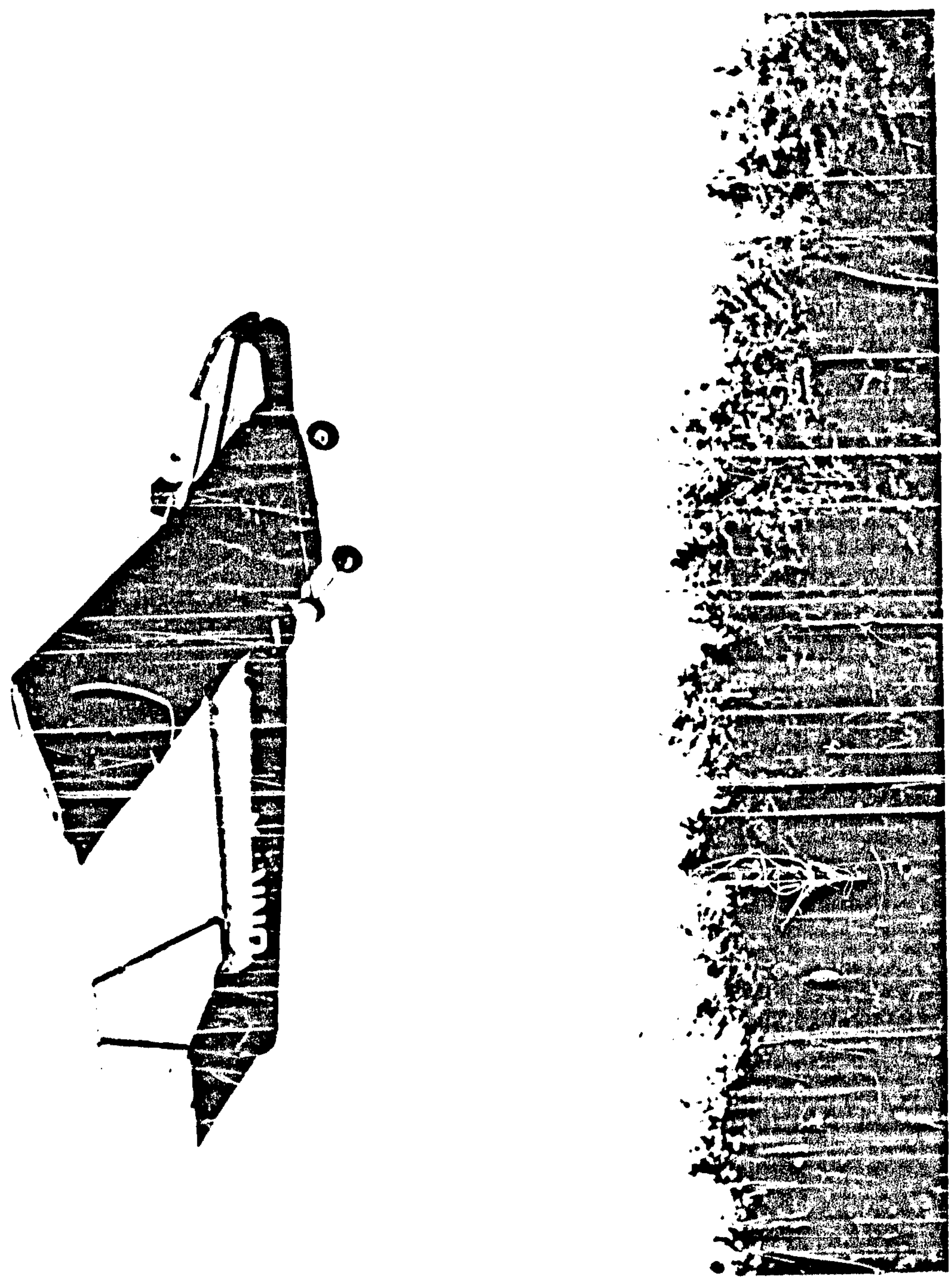

Figure 38. GA447 One-Place Inflatoplane Equipped with Tricycle Landing Gear (Optional)

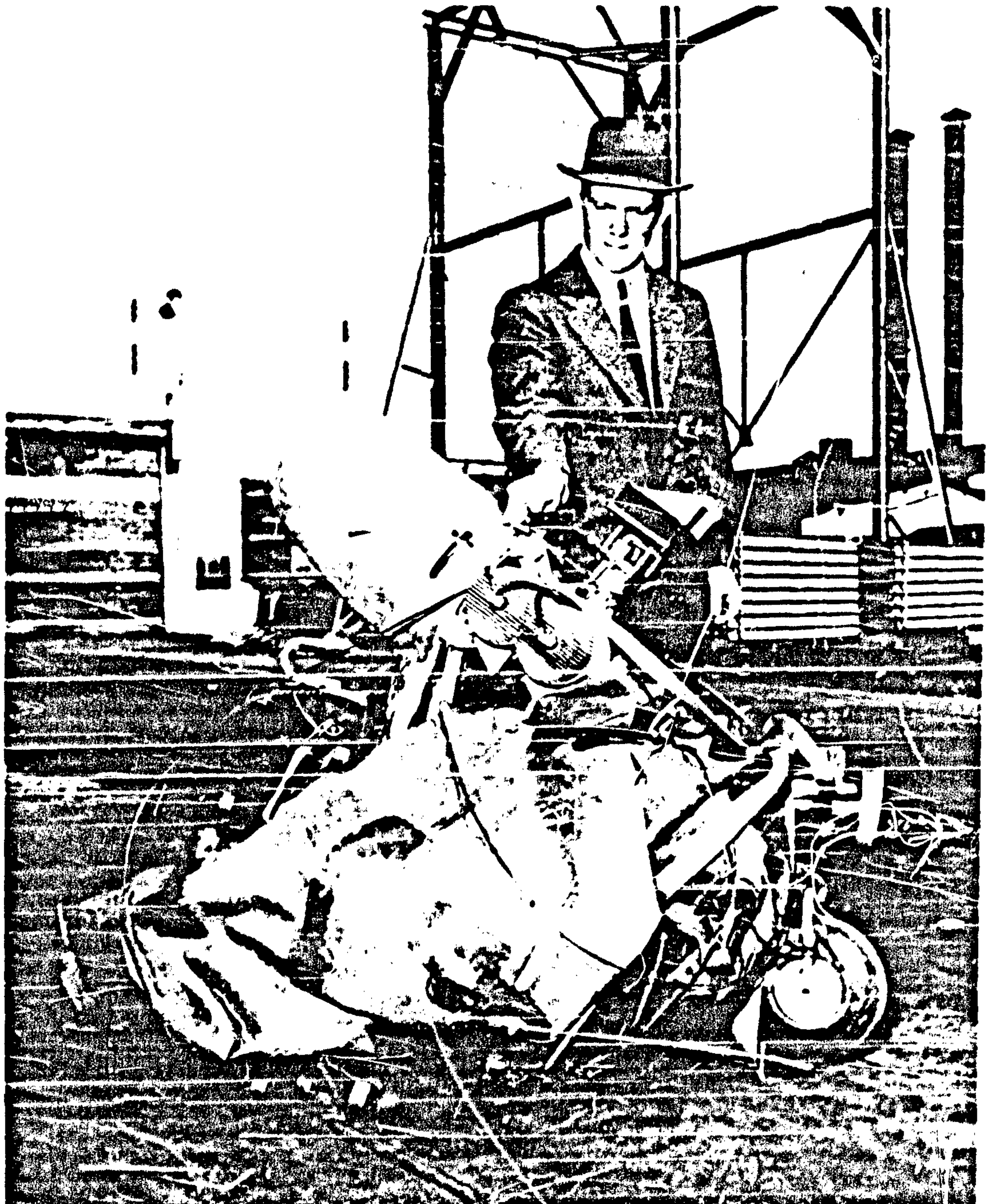

Figure 39. Stage I of Inflation Sequence. Inflatoplane is Removed from Container Uni

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Figure 40. Stage II of Inflation Sequence. The Inflatoplane is Spread Out on the Ground. Compressor Unit is then Actuated for the Proper Inflation Pressure

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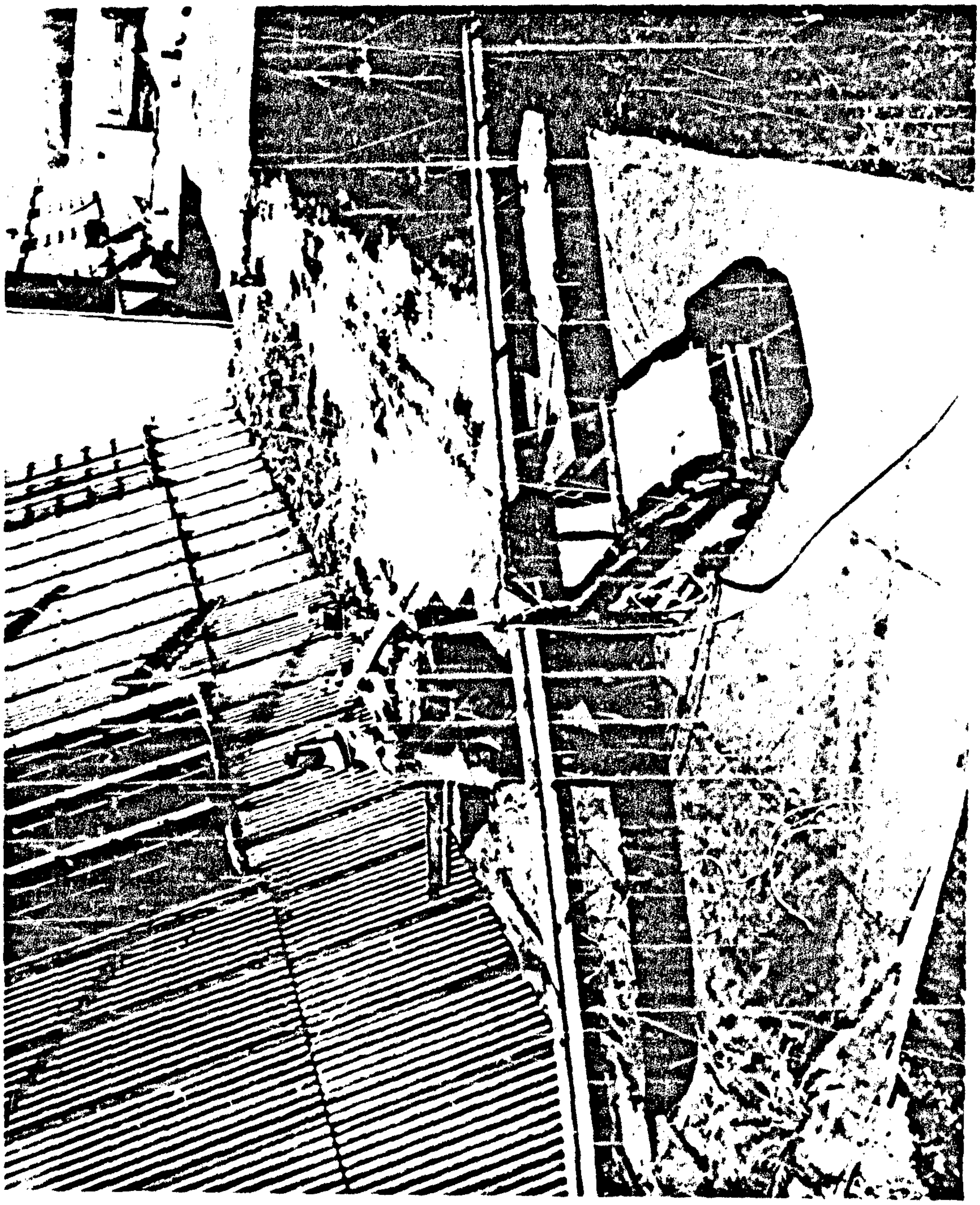

Figure 41. Stage III of Inflation Sequence. Plane is Now Inflated. Engine is Ready for Instantaneous Starting